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THE ENVIRONMENTAL TEST PROGRAM AND SYSTEMS EVALUATION OF THE S-3 ENERGETIC PARTICLES SATELLITE

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ENVIRONMENTAL TEST PROGRAM
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SUMMARY

The S-3 Energetic Particles Satellite was developed under the direction of Mr. Paul Butler and Dr. F. B. McDonald of NASA's Goddard Space Flight Center. The primary objective of the S-3 Satellite was to describe completely the trapped corpuscular radiation, solar particles, cosmic radiation, the solar winds, and to correlate particle phenomena with magnetic field observations.

Environmental testing of the prototype models and the Flight Spacecraft was performed by the Test and Evaluation Division of GSFC.

The major objectives of the test program were twofold: (1) to qualify the design of the spacecraft and its subsystems and (2) to demonstrate the dependability of flight hardware. The first objective was accomplished through Design Qualification tests of prototype items at levels of exposure more severe than those anticipated from handling, shipment, launch, and orbital flight. Acceptance tests of flight hardware, at expected environmental conditions, were conducted to achieve the second objective.

This report describes the test program, the test results, an evaluation of each spacecraft system, and presents conclusions and recommendations intended to initiate improvements in future spacecraft and their test programs.

Numerous difficulties were encountered during the test program, but none required extensive redesign or caused appreciable delays in the overall S-3 development program.

The Flight Spacecraft was successfully launched August 15, 1961, and was designated Explorer XII, as the Delta launch vehicle placed the spacecraft into its intended orbit.

Explorer XII ceased transmitting on December 6, 1961 after sending 2568 hours of real time data. This solar powered satellite had a useful orbital life of approximately four months.

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1. INTRODUCTION

This report presents a complete description of the Environmental Test Program of the S-3 Energetic Particles Satellite* and an evaluation of the spacecraft system and its test program.

The Environmental Test Program was designed to produce a high level of confidence in the ability of the S-3 Spacecraft (Figure 1) to successfully withstand the environments expected during handling, shipment, launch, and orbital flight.

The tests included in the Environmental Test Program were divided into two categories:

(1) Design Qualification Tests
(2) Acceptance Tests

The integrity of the mechanical and electrical design was determined by Design Qualification tests of a structural prototype and an operable prototype spacecraft unit. The dependability of flight items was established by acceptance tests of the Flight Spacecraft and the Flight Spares.

The basis for these tests and the methods by which they were carried out are presented in sections 2 through 4. The results of the tests can be found in section 5.

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* See Appendix A

Section 6 is devoted to an evaluation of each spacecraft model, while section 7 presents conclusions and recommendations intended to initiate improvements in future spacecraft and their environmental test programs.

Project development efforts culminated in a successful launch of the S-3 Flight Spacecraft (Figure 2) on August 15, 1961, from the AFMTC, Cape Canaveral, Florida.* The orbiting spacecraft (Figure 3) was designated "Explorer XII." The Delta launch vehicle (Figure 4) placed the spacecraft in a highly elliptical orbit ranging from 159 nautical miles at perigee to 41,764 nautical miles at apogee.

During 112 days in orbit, all experiments and instrumentation (Figure 5) functioned satisfactorily. Telemetry transmission ceased December 6, 1961.

2. BACKGROUND AND HISTORY

High reliability is a prime requirement for any effective space program. This requirement arises partially from the high cost of advanced and novel instruments characteristic of space research and the high cost of the launch vehicles. In addition, certain space operations must be successfully accomplished in a specific time period dependent on mission requirements. Thus, assurance of success must be enhanced by every possible means. An effective technique for obtaining such assurance is the application of laboratory tests which simulate, insofar as practical, the environmental conditions actually encountered by a spacecraft.

Such environmental test programs are planned to establish the dependability of a spacecraft under anticipated environmental conditions. A continuous long-term method for upgrading the environmental test programs is utilized in which data for future programs is provided by a thorough evaluation of previous spacecraft test programs and by comparing orbital performance data against results obtained during testing.

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* See Appendix B

This procedure (long term) incurs a time penalty for program improvement because of the necessity of waiting until after the spacecraft launch before a true evaluation of the spacecraft and its test program can be made. Despite this deficiency, this procedure is expected to lead to test programs resulting in a high degree of confidence in the ability of the spacecraft to survive the pre-launch and launch environments and to operate within design limits for the specified orbital lifetime.

Based on this philosophy, the Test Plan for the S-3 Structural Prototype*, dated October 3, 1960 (revised October 24, 1960), and the Environmental Test Plan, Energetic Particles Satellite, S-3**, dated November 29, 1960, were prepared specifying the tests deemed necessary, the test levels, test sequence, and certain monitoring procedures. Later, more refined estimates of anticipated environmental conditions and thermal predictions necessitated revisions to the Test Plan. All revisions and refinements were then incorporated into a single document (Acceptance Test Program for Flight Systems of the S-3 Energetic Particles Satellite***, dated May, 1961) for use in testing the Flight Spacecraft.

Testing of the Structural Prototype began October 8, 1960, but was suspended in order to modify the structural design. Structural Prototype testing was resumed November 3, 1960. Design Qualification Tests of the Prototype Spacecraft commenced on March 22, 1961, and continued into June, by which time fabrication of the Flight Spacecraft was completed and Acceptance Tests were started. The Flight Spacecraft was accepted as flight worthy in June, 1961, and Acceptance Testing of the Flight Spare Spacecraft began immediately and consumed about one month. The test operations were concluded with Acceptance Tests of a second set of spare subassemblies during July and August, 1961.

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* See Appendix C
** See Appendix D
*** See Appendix E

3. TEST OBJECTIVES

The S-3 Environmental Test Program objectives were:

(1) To verify that novel or unproven designs could successfully meet the required performance parameters and have satisfactory life expectancy under pre-launch and launch environments.

(2) To determine the ability of available hardware (proven in non-space applications) to withstand launch and space environments.

(3) To locate latent defects in material and workmanship, thereby providing assurance that none of the essential characteristics of the spacecraft had been degraded in manufacturing and accompanying inspection and handling.

Specifically, this program was designed to provide, within the limitations of the GSFC environmental test laboratory and the project schedule, the maximum possible assurance that the environmental stresses associated with spacecraft handling, shipment, launch, and orbital flight could be sustained.

4. TEST PLAN

To accomplish the objectives outlined above, each environmental exposure must be applied in accordance with a well-defined test program which gives proper attention to test levels, sequence of application, and detailed monitoring procedures.

The test program was composed of two major categories:

(1) Design Qualification Tests
(2) Acceptance Tests

Adequacy of the spacecraft structural design was determined by subjecting a Structural Prototype as well as the Prototype Unit to more severe levels of mechanical environmental stress than those expected from handling, shipment, and launch. These Design Qualification Tests increased the rate of random failures, enhanced the probability of detecting excessively high inherent failure rates, and revealed the "weak links in the chain" of subassemblies comprising the system. Test duration was selected to be sufficient to carry the item past the "infant mortality" stage. Design changes were made during the course of Design Qualification Tests, as necessary, and the item returned for retesting.

In Acceptance Tests, on the other hand, environmental levels representative of expected flight conditions were utilized to verify adequacy of spacecraft dependability. In this case, the design of the hardware had been previously tested and qualified, but was subject to defects in materials and workmanship.

Tests of the entire spacecraft system demonstrated the compatibility of all parts of the system under simulated launch and orbital environments.

To provide assurance against an excessive number of failures at the system level, testing at the subassembly level was proposed but left to the option of the experimenters and instrumentors.

The test plan was sufficiently comprehensive to provide all concerned with the test program with a basic knowledge of the mission requirements, the spacecraft and its functions, the method of testing planned, and the test sequence.

TEST DESCRIPTION

The tests employed were intended to simulate the various environmental conditions expected during preflight, launch, and orbit as described in the following paragraphs:

Balance

Static and dynamic balancing of the spacecraft was specified to ensure spin stability of the spacecraft during launch and orbital flight. Although not an environmental test, this procedure was a necessary prerequisite for the Flight Spacecraft. It was also specified for the Prototype Unit and Structural Prototype to develop techniques and procedures and to insure that these units would be dynamically similar to the Flight Spacecraft during mechanical testing (i.e., acceleration and vibration).

Spin Test

Prior to third-stage ignition, the spacecraft third-stage (X-248) combination is spun to 150 rpm (nominal). The spin test was intended to verify operation of the spacecraft system during this launch-phase condition.

Acceleration Test

The maximum acceleration (18 g), imparted to the S-3 Spacecraft by the Delta launch vehicle, occurs just prior to third-stage burnout. The orientation of the spacecraft on the centrifuge was selected so as to simulate the sustained loading of this thrust-induced acceleration. In addition, transverse acceleration tests were specified, based on expected handling loads of 2 g.

Shock Test

A shock environment is produced in several ways--handling, shipment, stage ignition, and stage separation being the most common. The S-3 shock test parameters were dictated by handling and transportation considerations, since the shock pulses generated by the Delta launch vehicle were expected to be less severe.

Vibration Test

Vibratory excitation of the spacecraft arises from shipment and rocket motor burning (primarily the X-248 third stage), as well as from acoustic and aerodynamic sources.

Sinusoidal and random exposures were conducted in three mutually orthogonal directions--parallel to the spacecraft spin axis and two arbitrarily selected, mutually perpendicular transverse axes.

Temperature Test

Two types of temperature tests were specified. The first test simulated temperatures which might be encountered under certain storage or transportation conditions. The second test attempted to simulate orbital temperature extremes, thus providing an early check of operability prior to attempting the more elaborate and time-consuming thermal-vacuum tests.

Humidity Test

The humidity test was designed to yield information regarding operability of the spacecraft during exposure to high relative humidity. Also of interest was the time required after exposure for any malfunctioning items to return to satisfactory operation.

Thermal-vacuum Test

This series of tests was intended to verify the operation of the spacecraft under the combined environments of temperature and high vacuum.

A (transmitter) thermal restraint on the attitude of the orbiting spacecraft dictated that the sun-spin axis angle must remain within ±60° of the spacecraft's equator* throughout the one-year lifetime. Thus, extreme heating or cooling of the transmitter would be

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* The spacecraft's equator refers to a plane perpendicular to the spin axis of the spacecraft.

avoided. Consequently, the thermal-vacuum tests were conducted by simulating the predicted thermal gradients (by using lamps) for a 30° solar aspect and for a 150° solar aspect (i.e., ±60° from the equator).

In addition, tests were specified at uniform hot and cold temperatures, representing the nominal orbital extremes of +35°C and -10°C, respectively.

5. TEST RESULTS

The tests were conducted, as noted above, in two distinct series (Design Qualification Tests and Acceptance Tests), in accordance with the test plans. The Design Qualification Tests were conducted on the Structural Prototype and the Prototype Unit, while the Acceptance Tests were conducted on the Flight Spacecraft and the Flight Spares.

Enumerated below are the results of these tests given in the order in which the tests were conducted:

STRUCTURAL PROTOTYPE

The Structural Prototype was tested during October and November, 1960.* The initial, low level, thrust axis vibration tests** (October 8, 1960) indicated that high amplifications were present on the instrument platform (Figure 6) at the primary structural resonance of 80 cps. Subsequent vibration tests, at qualification levels, caused the fracture of a stainless steel stud used to secure the simulated Cosmic Ray Logic Box to the fiber-glass-nylon honeycomb platform. This was remedied by using four studs of higher strength instead of the original two.

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* See Chart 1
** See Appendix F

The subsequent retest caused a failure of the instrument platform. The simulated Encoder Converter was torn from the platform during the sinusoidal exposure at the primary structural resonance. This necessitated repair and design modification of the platform support structure.

The payload was modified to include an aluminum ring that encircled the platform and furnished additional support from the underneath-side. The purpose of the ring was to stiffen the platform and prevent the diaphragming action which was responsible for both previous failures. Testing of the modified repaired Structural Prototype (designated Model "B") commenced on November 3, 1961. No further serious problem areas were detected as a result of the vibration, acceleration, or spin tests, and the structural design was considered to be qualified.

PROTOTYPE UNIT

The Design Qualification Tests of the Prototype Unit began on March 22, 1961, and ended on June 27, 1961. During this 98-day period, 60 days were consumed by actual tests, test preparations, repairs, or other work directly related to the tests.*

During the test program, approximately 37 discrepancies were encountered--16 of which were subassembly failures, one was marginal subassembly operation, and 12 were questionable subassembly operation. The remaining nine were ascribed to instrumentation, procedural, or facility difficulties.

Two of the 16 subassembly failures** could have resulted in mission failure had they occurred during flight. Of the remaining 14 failures, 11 would have caused loss of data from the affected experiment; the remaining three could have caused at least a partial mission failure, but only under an improbable simultaneous combination of conditions.

The following is an account of the results and the major discrepancies encountered during each test:***

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* See Chart 1

** See Chart 2

*** See Appendix G

Balance

The balancing operations (Figure 7) were somewhat hampered because of a high initial static unbalance. This unbalance was reduced by the addition of a lead plate which, because of the pressing test schedule, was not removed and resolved into components in each of the two correction planes. Hence, a minimum amount of ballast was not used. The results were as follows:

	Static	Dynamic
Initial Unbalance	430 oz-in	1840 oz-in^2
Residual Unbalance	7.5 oz-in	91.2 oz-in^2
Weight Added	2.57 pounds (non-magnetic lead)	

Spin Test

No major difficulties were encountered.

Acceleration Tests

No major difficulties were encountered. (Figure 8)

Vibration Tests

During the initial vibration test (sinusoidal, thrust axis - Figure 9), the Transmitter (previously successfully tested at qualification test levels for vibration and acceleration) and the Ion Electron Detector malfunctioned. The final amplifier tube of the Transmitter appeared to have a broken weld, while the photomultiplier tube of the Ion Electron Detector had malfunctioned. Both units were repaired and the test repeated. During subsequent vibration tests, the Regulator Converter and the Ames Proton Analyzer acted somewhat erratically, although post-test checkout indicated acceptable operation of both units. Also, one of the two "pancake" geiger tubes of the GM Telescope malfunctioned. Since no replacement tube was available, the geiger tube leads were interchanged so that operation of the satisfactory tube could be monitored during subsequent tests.* (Figure 10)

Temperature Tests

No malfunctions of spacecraft equipment occurred as a result of this test (Figure 11). However, several time-consuming problems (primarily faulty test instrumentation) were encountered which had to be resolved before the tests could be conducted.**

Thermal-vacuum Tests

During the first high-temperature vacuum exposure (+35°C), malfunctions of the Solar Array Voltage Regulator, the Single Crystal Detector, the Pulse Height Analyzer, and the Magnetometer Electronics occurred. In addition, excessive heating of the Regulator Converter necessitated modifications to the test procedures in order to prevent a malfunction. This problem was ameliorated (as verified by subsequent tests) by the addition of a conductive heat radiator in contact with the Regulator Converter.

The failure of the Solar Array Voltage Regulator, apparently because of thermal runaway of a transistor, shorted the Batteries, thereby burning out the Current Sensor. This condition isolated the spacecraft electronics from the Solar Array Power Supply. The Solar Array Voltage Regulator was redesigned to include better heat dissipation and an increase of power capabilities. The Single Crystal Detector (failure of the high voltage power supply caused by extensive arcing and corona discharge) was replaced by a new unit. The Pulse Height Analyzer contained a cold solder joint, while the Magnetometer Electronics was repaired by changing two amplifier transistors after having operated satisfactorily in the subsequent test.

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* See Appendix H
** See Appendix J

During the first low-temperature vacuum soak (-10°C), the final stage of the Transmitter failed to start when the temperature of the unit was less than -7°C, although normal operation was observed at higher temperatures. A starting resistor in the Transmitter Converter was changed. The second "cold-soak" exposure again indicated a failure of the final stage to start properly (temperature, -10°C). However, subsequent thermal-vacuum tests of the Transmitter at 0°C, -10°C, and -20°C failed to cause duplication of this discrepancy.

Also, during the first cold-soak exposure, the Program Switch failed to respond to command. During subsequent testing, questionable and/or marginal operation of the Program Switch and Recycle Timer was observed.

The second hot and cold-soak exposures were conducted without incident, except for the loss of readout from the Pulse Height Analyzer (defective transistor in decoding gate) and the other Program Switch-Recycle Timer discrepancies noted above.

At the outset of the 45° solar aspect exposure (Figure 12), a malfunction occurred in the GM Omni-directional Counter (transistor failure in the converter primary). The GM Omni-directional Counter was replaced with a new unit and the test restarted. A failure occurred in the power supply of the Double Telescope, but no corrective action was taken at this time.

A final hot-soak exposure did not disclose any additional discrepancies.*

Shock Test

No difficulties were encountered.

Humidity Test

The performance of several subassemblies was degraded (as expected) during the exposure, but recovered shortly after the conclusion of the test.

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* See Appendix K and Table 1

Prototype Unit Retest

Because of the many discrepancies which occurred during the test program, a decision was made to subject the Prototype Unit to vibration and vacuum retests. It was felt that these tests could be performed within a few days, commensurate with the pressing schedule required to meet the launch date and, if successful, would demonstrate that at least the minimum requirements for design qualification could be met.

Accordingly, sinusoidal and random vibration tests were conducted on June 24 and 25, 1961. The GM Telescope failed during the shaker system equalization procedure* after having passed the sinusoidal test satisfactorily. No other difficulties were encountered.

The two-day vacuum test (June 25 through 27, 1961), at ambient temperature, was conducted with satisfactory results.

It was concluded that the prototype design was qualified, but that separate Design Qualification Tests should be conducted on the GM Telescope after design modifications were accomplished.

FLIGHT SPACECRAFT

Acceptance Tests of the Flight Spacecraft were initiated on May 20, 1961, and continued through June 13, 1961. During this 25-day period, 21 days were consumed by actual tests or set-up operations (four days - balance, one day - vibration, and 16 days - thermal-vacuum).**

During the test program, no critical failures occurred.

The following is a brief account of the results and the discrepancies encountered during each test:***

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* See Appendix H, page H-3.

** See Chart 1

*** See Appendix L

Balance

As with the Prototype Unit, the Flight Spacecraft exhibited a relatively high initial static unbalance. In addition, the low balancing speed (150 rpm)*, the payload mass configuration, and balancing machine difficulties accounted for a short delay in the completion of the operation. The results were as follows:

	Static	Dynamic
Initial Unbalance	426 oz-in	2340 oz-in^2
Residual Unbalance	8 oz-in	18 oz-in^2
Weight Added	2.27 pounds (non-magnetic lead)	

Vibration Tests

No difficulties were encountered.**

Thermal-vacuum Test

Prior to the test, it was discovered that one of the two photomultiplier tubes of the Double Telescope had burned-out. A spare Double Telescope was successfully vibrated at Acceptance Test levels and installed in the spacecraft. The power supply of the GM Telescope malfunctioned during chamber evacuation; a spare GM Telescope was vibrated at Acceptance Test levels, but malfunctioned in a separate vacuum test. After repairs, this GM Telescope again malfunctioned during a separate vacuum retest, but was repaired and installed in the spacecraft.

However, the GM Telescope malfunctioned again when the coincidence readout was lost during the 150° Solar Aspect exposure (apparently due to a failure of one of the "pancake" geiger tubes).***

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* The next highest operating speed of the Trebel balancing machine, 225 rpm, approached the structural design limits of the spacecraft.

** See Appendix M

*** See Chart 2 and Appendix N

Final Balance

On July 8, 1961, prior to shipment to Cape Canaveral, the Flight Spacecraft received a final "Flight Balance." The results were as follows:

	Static	Dynamic
Initial Unbalance	16.7 oz-in	76.25 oz-in^2
Residual Unbalance	1.15 oz-in	6.75 oz-in^2
Weight Added	40 grams (Total Balance Weight: 2.36 lb)	
Total Weight of Spacecraft	83.6 lb	

FLIGHT SPARE SPACECRAFT

The Acceptance Tests of the Flight Spare Spacecraft took place during the period June 19, 1961, through July 7, 1961--a total of 19 days, 18 of which were consumed by actual tests or test preparations (balance, static only - 1-1/2 days, vibration - 2-1/2 days, thermal-vacuum - 14 days).*

The following is an account of the results and the major discrepancies encountered during each test:**

Balance

In view of the pressing schedule, only static balancing of this spacecraft was conducted at this time, since a complete balancing operation would be performed shortly before shipment of the unit to Cape Canaveral. Balancing problems encountered were similar to those noted for the Flight Spacecraft. The results were as follows:

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* See Chart 1

** See Appendix L

	Static
Initial Unbalance	381 oz-in
Residual Unbalance	6.5 oz-in
Weight Added	1.98 lbs. (non-magnetic lead)

Vibration Tests

The Calidyne 177A Vibration System was used to test the Flight Spare Spacecraft since modifications to the MB-C50 System, which had been utilized for all previous S-3 spacecraft vibration testing, were underway. The higher magnetic field of the 177A severely disrupted the operation of the Telemetry Encoder, thus preventing adequate data analysis. Attempts to reduce the effect of the field of the Telemetry Encoder were unsuccessful, and it was necessary to conduct the tests despite the interference. The GM Telescope failed during the random-thrust axis vibration test. The operation of the GM Telescope became intermittent during subsequent vibration tests.*

Thermal-vacuum Test

The Transmitter malfunctioned at the initial checkout of the 30° solar aspect exposure. The amplification stage was found to be misaligned and required retuning. The second 30° solar aspect exposure pointed out an incompatibility between the Batteries at -10°C and the undervoltage lockout feature of the Program Switch. Investigation revealed that the normal operating plateau of the Batteries at -10°C was less than the undervoltage lockout setting. The Program Switch was adjusted so that undervoltage lockout would occur at a lower voltage than the battery voltage. During the -10°C cold-soak exposure, a short

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* See Appendix M

circuit developed in the test instrumentation internal to the chamber which prevented checkout of the Double Telescope, the Pulse Height Analyzer, and the Single Crystal Detector. After the completion of the test, the chamber was opened, the difficulty repaired, and a short (12-hour) test at -10°C was conducted to verify the operation of the above-mentioned units. Inspection of the Ion-Electron Detector (which exhibited intermittent operation during the 150° solar aspect exposure), after the completion of the tests, indicated that the unit had been damaged by a screw which had worked loose from the absorber wheel motor mount.*

Final Balance

The Flight Spare Spacecraft was balanced on July 13, 1961, just prior to shipment to Cape Canaveral.** The results were:

	Static	Dynamic
Initial Unbalance	16.4 oz-in	29.6 oz-in^2
Residual Unbalance	1.6 oz-in	6.2 oz-in^2
Weight Added	39 grams (Total Balance Weight: 2.07 lb)	
Total Weight of Spacecraft using Flt. Solar Paddles	83.2 lb	

SUBASSEMBLY TESTS

Approximately 40 different subassemblies and 22 components (tubes, transistors, etc.) were subjected to a total of 76 and 22 tests, respectively.***

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* See Appendix N
** See Appendix P
*** See Appendix Q

Except for the prototype Transmitter, no Design Qualification Tests or Acceptance Tests were conducted on S-3 subassemblies prior to the tests they received as an integral part of the respective systems.* However, the spacecraft Design Qualification Test and Acceptance Test operations, as well as post-test failures, led to the necessity of testing, on a per case basis, of several other subassemblies. Specifically, the GM Telescope received separate Design Qualification Tests, while the Transmitter**, Ion-Electron Detector (Flight Unit and Flight Spare), GM Telescope, and several Flight Spare #2 subassemblies received separate Acceptance Tests. These separate subassembly Acceptance Tests were limited to vibration tests at levels based on transmissibility data and a short duration thermal-vacuum test at 0°C and +40°C.

6. SYSTEM EVALUATION

The purpose of this section of the report is to discuss and analyze the test results from the system point of view. Each system is discussed separately except where interrelated problem areas appear.

No attempt is made to analyze circuit design, or the use of components, or to assign a numerical reliability factor, either at the subassembly or system level. However, the material presented may serve to point out weaknesses uncovered by the tests, which are worthy of a detailed study and analysis in order to improve future designs.

STRUCTURAL PROTOTYPE

Accelerometer recordings made during the vibration tests of this unit revealed that moderate amplifications were present on the instrument platform. Thus, the vibration environment experienced by the simulated experiments within the resonant frequency range of the platform

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* However, each of the experimenters and instrumenters conducted tests (primarily temperature) prior to Systems Test.

** See Appendix R, Part 2

was higher than the input excitation. These amplifications ranged from about 5:1 to about 10:1* at most locations around the platform within a frequency range of approximately 80±30 cps.

The amplifications (which could be detrimental to the experiments, since they were to be mounted to this platform) were lessened by the addition of a stiff aluminum ring encircling the platform and supporting it from underneath. However, this aluminum ring was deleted from subsequent spacecraft, presumably because of weight considerations.

Though several failures of electronic packages on the platform occurred during Prototype Unit and Flight Spare vibration tests, retests of these units, as well as the tests of the Flight Spacecraft and the launching itself, substantiated the fact that the electronics and associated hardware could successfully withstand the amplified vibrations.

PROTOTYPE UNIT

During the Qualification Tests of the Prototype Unit, sixteen failures occurred which, had they occurred in flight, could have resulted in either a total or partial mission failure, or a total or partial loss of data from an individual experiment.**

Two potential "catastrophic" failures--those resulting in possible mission failure, were the loss of the final amplifier stage of the Transmitter during vibration and the failure of the Solar Array Voltage Regulator during the (hot-soak) thermal-vacuum exposure.

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* These data are taken from oscillograph recordings of the accelerometer signals. Analysis of the harmonic components in the output waveform was not possible at the time, but would probably reveal distortion of the waveform with higher harmonics, causing erroneous interpretation of the oscillograph data.

** See Chart 2

The failure (at approximately 1000 cps) of the final amplifier vacuum tube (Sylvania 5977) of the Transmitter was attributed to a broken weld. The effect of the failure was to decrease the power output to approximately 100 milliwatts from the normal 1.5 watts.

This particular Transmitter previously had successfully passed the subassembly qualification vibration test. This test was conducted using the identical input accelerations which were applied to the Prototype Unit at the time of failure. Also, since the Transmitter is located within the center tube just above the spacecraft third stage interface, it is reasonable to assume that little or no amplification of vibration was transmitted to the unit during the system test. If this was the case, then failure occurred (sinusoidal vibration, thrust axis) at a level of vibration which did not exceed that which it had previously withstood. Microscopic examination of the broken weld indicated a possibility that the weld was previously defective. It is also possible that mechanical vibrations at qualification test levels, combined with the weakening of the cathode strap-to-cathode weld, because of frequent on-off cycling of the transmitter, could have caused the failure.

Each tube was examined microscopically prior to use in subsequent Transmitter units. Retest (same unit, tube replaced) at Qualification Test levels, and subsequent tests of the Flight Spacecraft, did not cause failure.

The second "potential catastrophic" failure occurred during the first thermal-vacuum hot test (+35°C) and involved the malfunction of the Solar Array Voltage Regulator. This unit (which regulates the Solar Paddle voltage and dissipates any excess power from the Solar Paddles) malfunctioned as a result of thermal runaway of the power transistor. The resulting lowered impedance of the Solar Regulator permitted discharge of the Batteries through the Current Sensor in the reverse direction, burning out the coils of the Current Sensor between the Solar Paddles and the Batteries.

This open circuit would have prevented charging of the Batteries as well as operation of the spacecraft from the Solar Paddles (Figure 13). The Solar Regulator was redesigned to include a parallel combination of two power transistors, each in series with a resistor, as well as improved heat dissipation capabilities (by mounting the transistors to separate structural struts). Subsequent tests indicated that the redesigned Solar Regulator adequately dissipated the excess power.

Other significant discrepancies which occurred during Qualification Tests included the failure of the Transmitter Converter to start at low temperatures, overheating of the Regulator Converter, and the failure of the undervoltage lock-out circuit in the Program Switch.

Each of these, though not as potentially disastrous as those previously mentioned, could have caused at least a partial failure of the spacecraft had they occurred in flight. However, mitigating circumstances surround each of these events. Before any of these discrepancies could have caused a loss of data from the spacecraft, an unlikely (though not impossible) combination of two or more conditions would have had to exist before the discrepancy could occur.

Starting the Transmitter when at a low temperature implies that the spacecraft had been previously turned off due to low power output from the Solar Paddles (or other causes) and that the sun/spin axis angle was small enough so that the Transmitter temperature was near its lowest expected extreme.

Improvement in the cold temperature starting characteristics of the Transmitter Converter was obtained by changing a starting resistor.

Overheating of the Regulator Converter (temperature reached +59.5°C and although still rising had begun to level off--maximum capability of the device was quoted as +60°C) occurred during the hot test (+35°C) with an input overvoltage to the Regulator Converter of 15 volts. The overheating was reduced by improving the conductive paths from the Regulator Converter and by the application of a black coating to each subassembly within the instrument compartment. Heat dissipation was substantially improved (lowered temperature by approximately 8°C) as indicated by subsequent tests.

During the thermal-vacuum hot-soak test, a condition occurred which pointed out the inability of the Program Switch to turn off the spacecraft when the rate of change of voltage was rapid.

(The Program Switch incorporated an undervoltage lock-out circuit which was designed to turn the spacecraft off when the Battery voltage was less than 12.8 volts. The spacecraft would then go through an eight-hour recycle period, during which time the Solar Paddles would be able to recharge the Batteries.)

This portion of the hot-soak test was conducted at an operating voltage of 13 volts to ascertain the current drain and temperature distribution within the spacecraft at the lower voltage. The external power source was supplying all the power to operate the spacecraft, while the Batteries were almost completely discharged. While operating at these conditions, an accidental momentary interruption of the power source transferred the entire load to the Batteries, causing the battery voltage to drop rapidly through the lock-out region, but the Program Switch failed to turn the spacecraft off.

Although the test conditions surrounding this "failure" were unrealistic, it was decided that condenser storage added to the undervoltage lock-out circuitry would eliminate this problem. Subsequent tests after modifications proved satisfactory operation.

Several scientific experiment packages within the Prototype Unit also malfunctioned during Qualification Tests. The GM Telescope failed on three occasions; the Pulse Height Analyzer and GM Counter twice; while the Single Crystal Detector, Double Telescope, Ion-Electron Detector and Magnetometer (electronics) each failed once.

Most of the problems associated with the GM Telescope were due to design, while the other experiments suffered from defective components or poor quality control. For example, the initial Pulse Height Analyzer units contained a very high percentage of cold-solder joints, although most of them were detected and resoldered prior to testing. Later models showed greatly improved workmanship.

FLIGHT SPACECRAFT

The Acceptance Tests of the Flight Spacecraft were accomplished with a minimum of trouble. Malfunctions of the Double Telescope (ascribed to a random failure of one of the photomultiplier tubes during checkout) and the GM Telescope (two failures - the first, during chamber evacuation, was traced to a burned-out power supply; the second, while in the 150° solar aspect thermal-vacuum exposure, was traced to a malfunction of one geiger tube) were the only failures that occurred during testing.

The spacecraft performed well during all tests, indicating that the solutions to the problems encountered with the Prototype Unit were adequate--at least under expected flight conditions.

However, several of the events that occurred during the pre-launch operations at Cape Canaveral* somewhat dampened the hopes for a completely successful spacecraft mission. The problems associated with the excessive number of operations performed on the spacecraft (partial disassembly, assembly, etc.), the inadequacy of the despin circuitry, the newly discovered idiosyncracies of the Program Switch, as well as the Transmitter discrepancy, all contributed toward reducing the level of confidence which had been obtained during the Acceptance Tests. However, it should be realized that the recognition of the incidental FM characteristic of the Transmitter permitted the replacement of this unit and thereby greatly improved the chances of a successful mission by eliminating the data reduction problems that might have otherwise ensued.

FLIGHT SPARE SPACECRAFT

Acceptance Tests of the Flight Spare Spacecraft were accomplished without major difficulty. The performance of the spacecraft during the vibration test was as good as that of the Flight Spacecraft except for the failure of the GM Telescope. During the thermal-vacuum test, operation of the spacecraft was marred by an incompatibility between the Program Switch and the Batteries, in addition to a detuned Transmitter and damaged tube of the Ion-Electron Detector.

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* See Appendix R, Part 2

The Program Switch-Battery difficulty was due to the fact that the operating voltage level of the Batteries at $-10^{\circ}C$ was less than the undervoltage lock-out setting of the Program Switch. The solution to the problem (adjustment of the lock-out setting to a low enough level (12.0 V) to permit operation was satisfactory for the purposes of completing the tests of the spacecraft, especially since no replacement batteries were available.

The damaged phosphor coating in the Ion-Electron Detector tube was the result of a loose screw from the absorber wheel motor mount. The physical dimensions and characteristics of the experiment were such that recoating the tube was not possible. Consequently, design modifications to the experiment were accomplished by the experimenter.

7. CONCLUSIONS AND RECOMMENDATIONS

It is believed that study and application of the following conclusions and recommendations could lead to the improvement of future spacecraft designs and/or test programs.

SPACECRAFT STRUCTURE

The honeycomb instrument platform exhibited amplifications of the vibration inputs within its resonant frequency range (80 ±30 cps). While the electronic packages on the platform generally withstood these amplifications, increased system reliability would result if these amplifications were reduced. Structural and dynamic analyses of the design of instrument platforms could lead to configurations with reduced amplifications at structural resonance.

TRANSMITTER

The failure of the Transmitter tube (Prototype Unit vibration test) could have been due to a defective weld, fatigue, or a random failure.

Substitution of a solid state transmitter (as presently being developed) would increase dependability since: (1) it replaces the final amplifier vacuum tube with transistors which are less susceptable to damage from vibration and (2) it eliminates the necessity for the Transmitter Converter (which caused several problems during the test program).

The net result of replacing the vacuum tube with transistors and the associated elimination of components is to nearly double the calculated mean time between failures for the Transmitter.

TEST POINTS

A practical problem inherent in the S-3 configuration, which manifested itself during the test program and launch site activities, was the inaccessibility of most of the electronic modules. Calibration or removal of packages in the spacecraft usually required removal of the top and bottom covers (a time-consuming process--especially during thermal-vacuum testing). In some cases, removal of a defective module necessitated the removal of other packages.

It would be advantageous if provisions for external test and calibration points for each package could be incorporated into future spacecraft designs. In addition, removal of individual units, without dismantling portions of the spacecraft, would save considerable time.

IDENTIFICATION

Several subsystems received for test bore little or no identification. During the course of the testing, many subsystems were switched from one spacecraft to another, or were replaced with new units. Thus, those which possessed no identification or those which were labeled "Flight Spare," etc., and were installed in the Flight Spacecraft, caused considerable confusion.

All subsystems should bear a permanent identification in a prominent location, so that they can be readily identified when installed in a spacecraft. Identification should be by means of a part or drawing number and a serial number and not by the designations "Prototype," "Flight," "Spare," etc. Also, major components within a subsystem should be identified on the outer package. This should include such things as photomultiplier tubes, geiger counters, or other items which might be substituted quite readily.

DESIGN REVIEW

The failure of the Solar Array Voltage Regulator (Prototype Unit thermal-vacuum tests) was due to design weaknesses and points out the need for careful analysis of the anticipated power dissipation requirements and the accompanying thermal considerations in the design of such devices.

A comprehensive and critical review of the entire spacecraft design, as well as a reliability study, should be conducted prior to the Qualification Tests. Any designs discovered to be deficient through this review should be redesigned and the new designs incorporated in the Prototype for testing. When this has been accomplished, all designs should be frozen.

ACCEPTANCE TESTS

All malfunctions of the Prototype Unit occurred during vibration and thermal-vacuum testing or at ambient conditions during spacecraft checkout. This result lends practical support to the theory that Acceptance Tests of Flight Spacecraft should include vibration and thermal-vacuum tests as a minimum.

TEMPERATURE TESTS

The operational temperature tests (Prototype Unit) failed to accomplish their purpose (finding thermal problems in the system easily and quickly, thereby saving time and effort from subsequent thermal-vacuum tests). Seventy-five percent of all major discrepancies occurred in thermal-vacuum, yet, not one had been detected during temperature tests. Therefore, no time or effort was saved. On the other hand, the practice of temperature testing a payload <u>prior to the start of the test program</u> (preliminary system temperature test), while many of the subassemblies are unpotted, proved to be extremely advantageous. This proved to be an effective time saver and served to eliminate many obvious deficiencies early in the program.

TESTS OF THERMAL DESIGN

The test programs did not encompass testing of the thermal design of the spacecraft. This was not by choice, but because of the lack of facilities to properly test and evaluate the design. Using future facilities, the thermal design should be tested as an integral part of future spacecraft designs. Also, the fabrication and testing of a thermal prototype early in a program would be invaluable, serving to eliminate many of the thermal problems encountered during prototype thermal-vacuum tests.

ORBITAL CONFIGURATIONS SPIN TEST

The S-3 test program did not include an "orbital configuration" spin test. This should be specified for future programs--especially those utilizing solar cell power supplies. The test should include the monitoring of the spacecraft as it is spun (at the intended orbital spin rate) and the "sun-angle" is swept through the predicted range.

SIMULATED PRE-LAUNCH OPERATIONS

Insofar as is practicable, the test program should include simulated "pre-launch operations." The Prototype Unit could be utilized to give program people (especially those who have not previously participated in pre-launch operations) a chance to familiarize themselves with the operations which take place at the launch site.*

MAGNETIC FACILITY

Present facilities did not permit calibration of the Magnetometer while in thermal-vacuum; as a result, the spacecraft were transported to Fredericksburg, Virginia, in order to accomplish calibration in a controlled magnetic field. A GSFC Magnetic Facility (as presently planned) would prove beneficial both to the experiment and schedules.

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* See Appendix R

REFERENCES

In addition to the documents incorporated in the appendixes, the following may be consulted for certain supplemental information:

1. Test Report - Subject: Structural Prototype, Energetic Particles Satellite, S-3, dated March 1, 1961

2. Memorandum Report - Subject: Acceleration Tests of S-3 Despin Timers, dated April 4, 1961

3. Test Report - Subject: Environmental Vibration Test, Prototype System, Energetic Particles Satellite, S-3 - 321.2(JC)S-3-06, dated May 1961

4. T&E Report No. 62-103 - Subject: Thermal-Vacuum Testing of the S-3 Prototype, Energetic Particles Satellite, dated August 8, 1961

5. T&E Report No. 62-104 - Subject: Vibration Tests of S-3 Flight Unit No. 1 and Flight Spare, dated August 18, 1961

6. T&E Report No. 62-106 - Subject: The Thermal-Vacuum Testing of the S-3 Energetic Particles Satellite Flight and Flight Spare Systems, dated September 13, 1961.

7. Memorandum Report No. 621-5 - Subject: Methods of Control and Evaluation of Mass Unbalance as Applied to the S-3 Energetic Particles Satellite, dated November 29, 1961

ILLUSTRATIONS, CHARTS, AND TABLES

List of Illustrations, Charts, and Tables

ILLUSTRATIONS

List of Illustrations, Charts, and Tables
(continued)

CHARTS

Chart No.

TABLE

Table No.

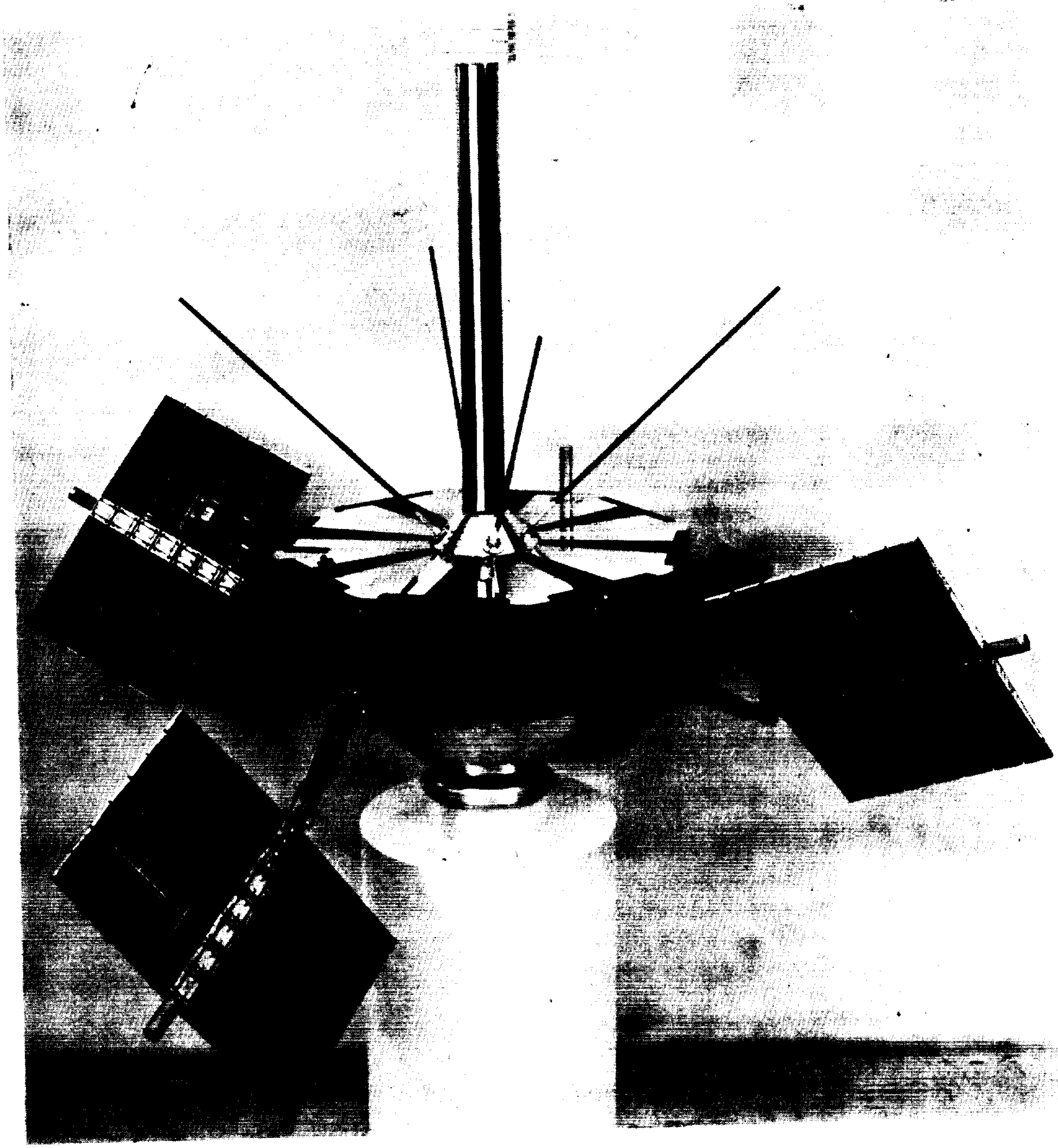


Figure 1. S-3 Energetic Particles Satellite

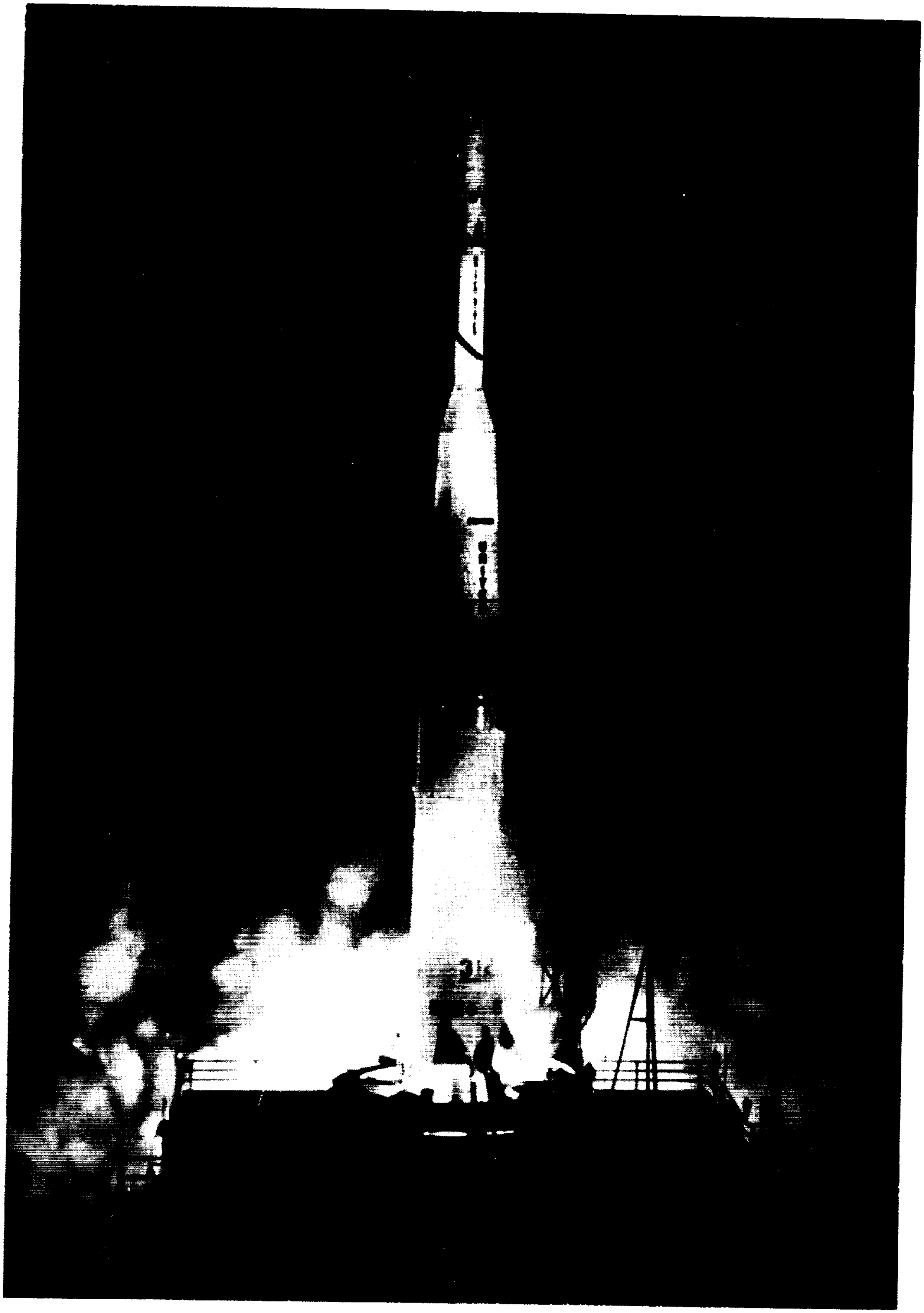


Figure 2. Launching of Explorer XII
August 15, 1961

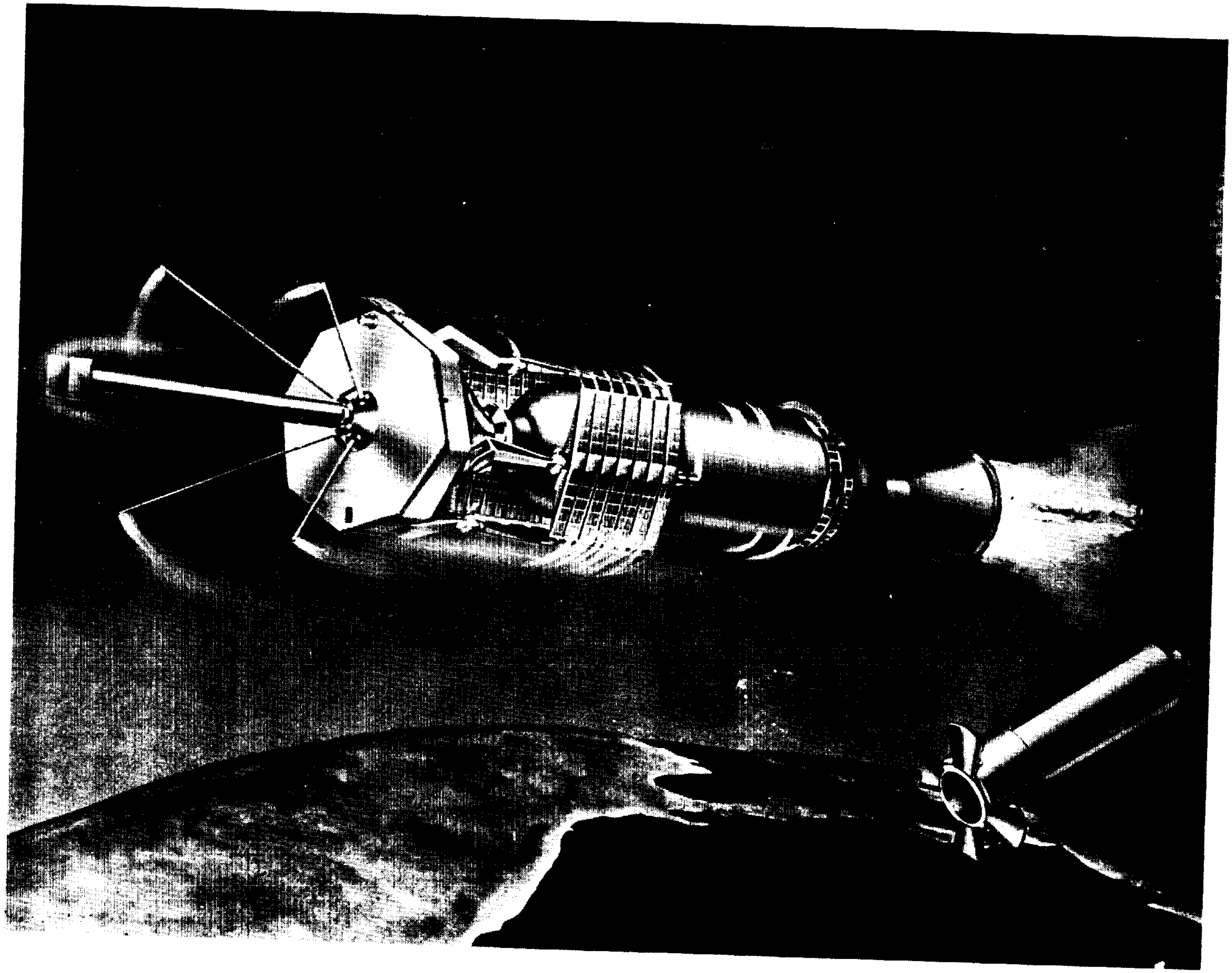


Figure 3. Injection of Explorer XII Into Orbit
(Artists Conception)

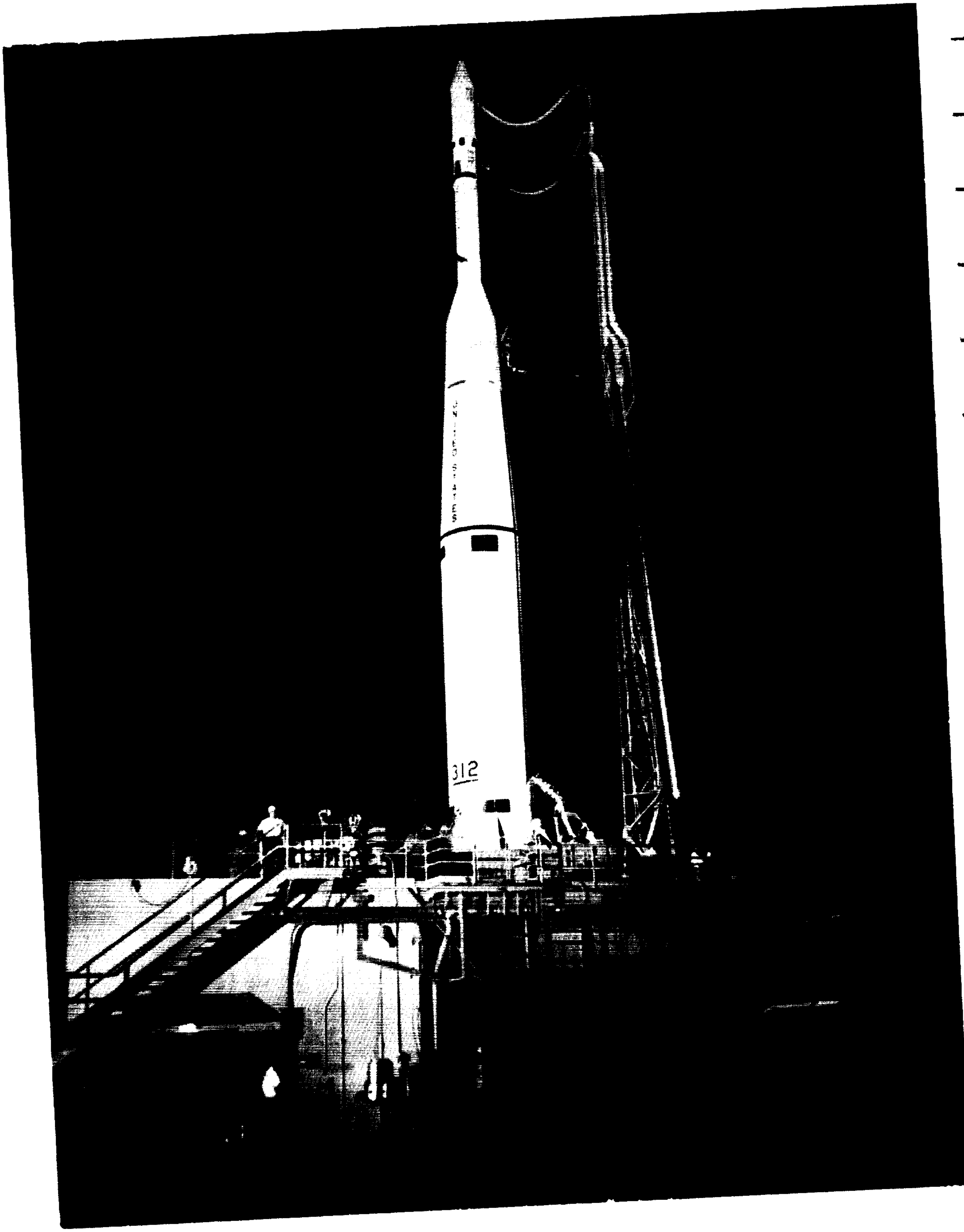


Figure 4. Delta Launch Vehicle

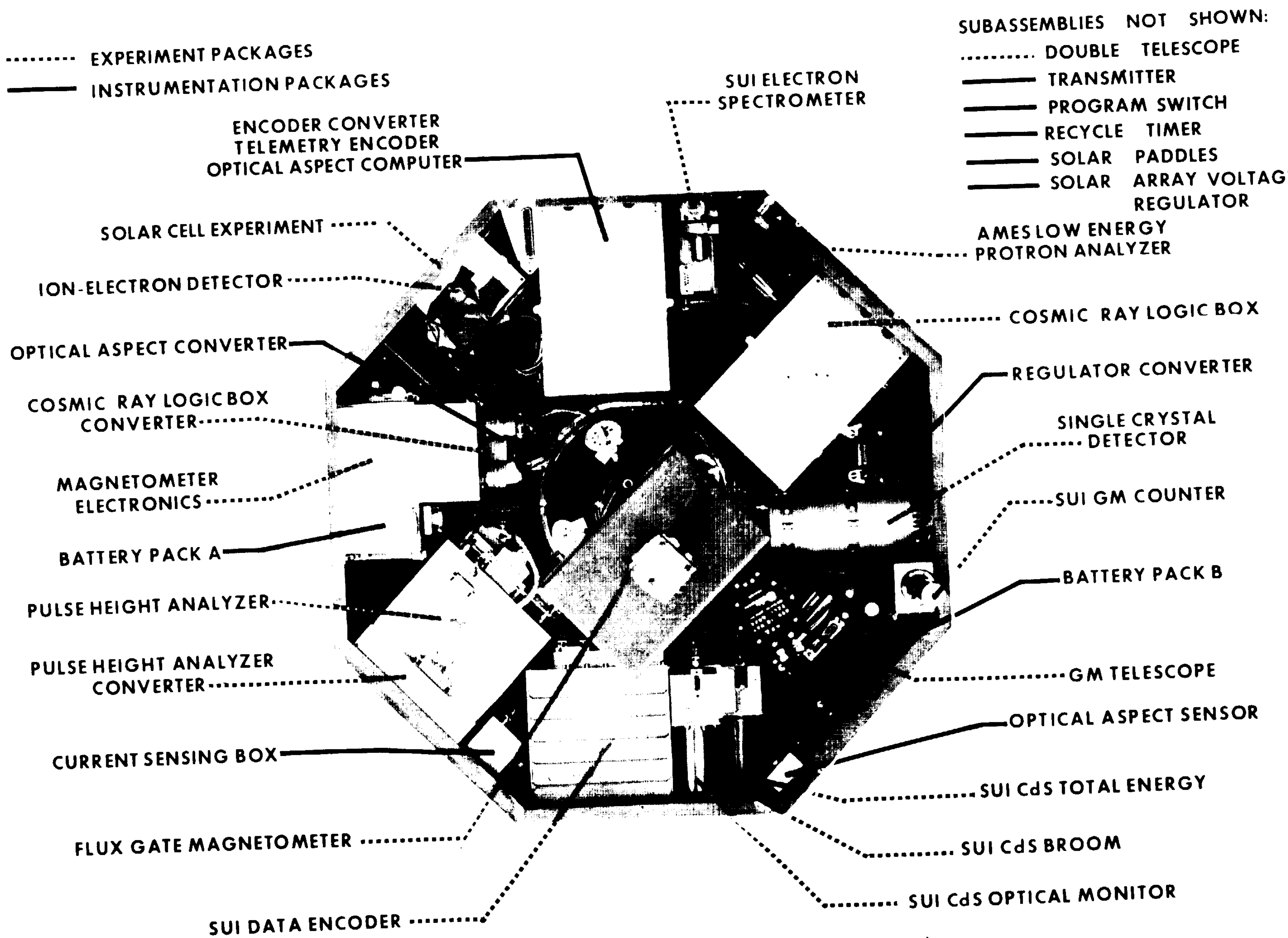


Figure 5. S-3 Experiments and Instrumentation (Prototype Unit)

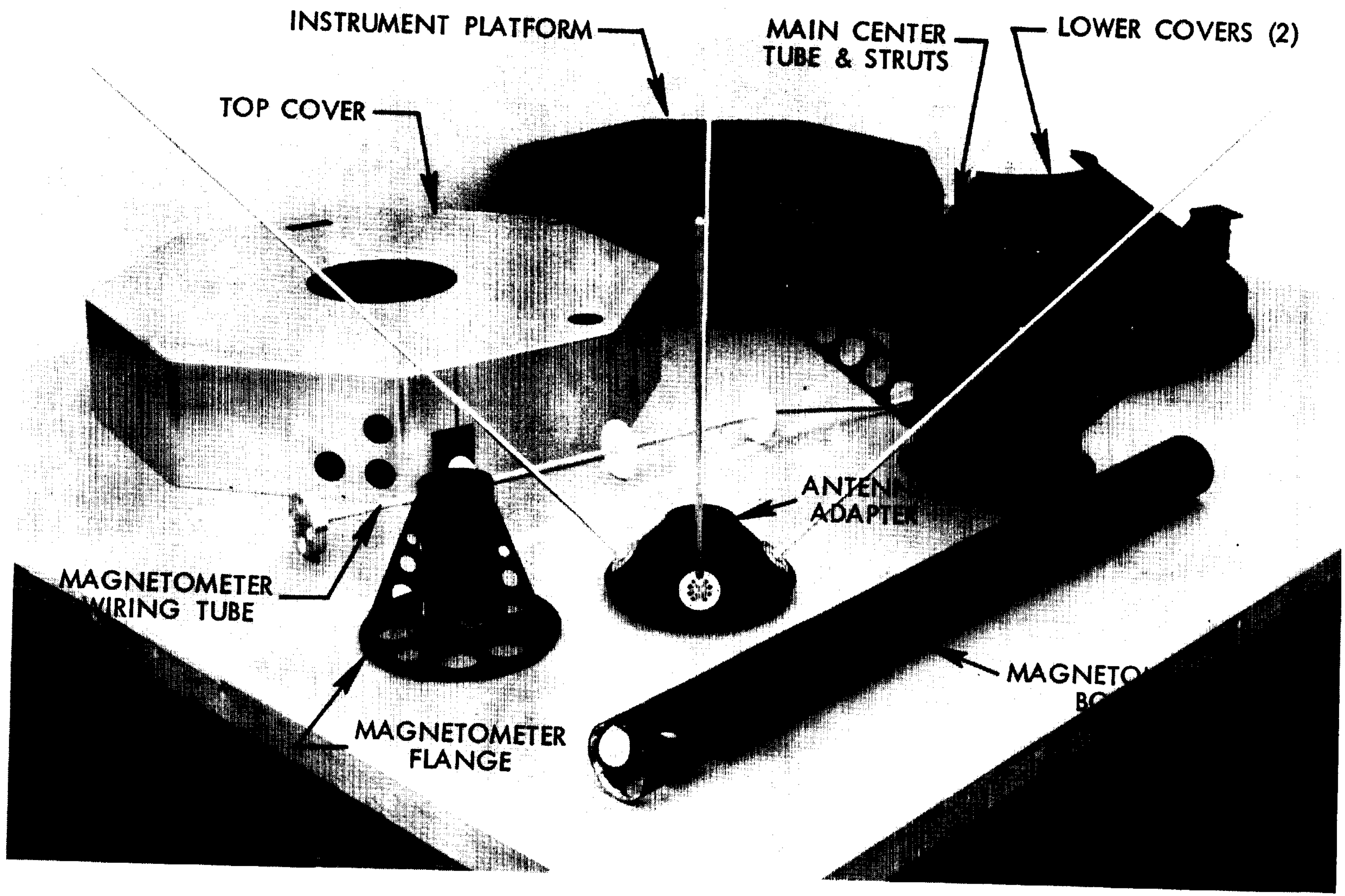


Figure 6. Structural Components

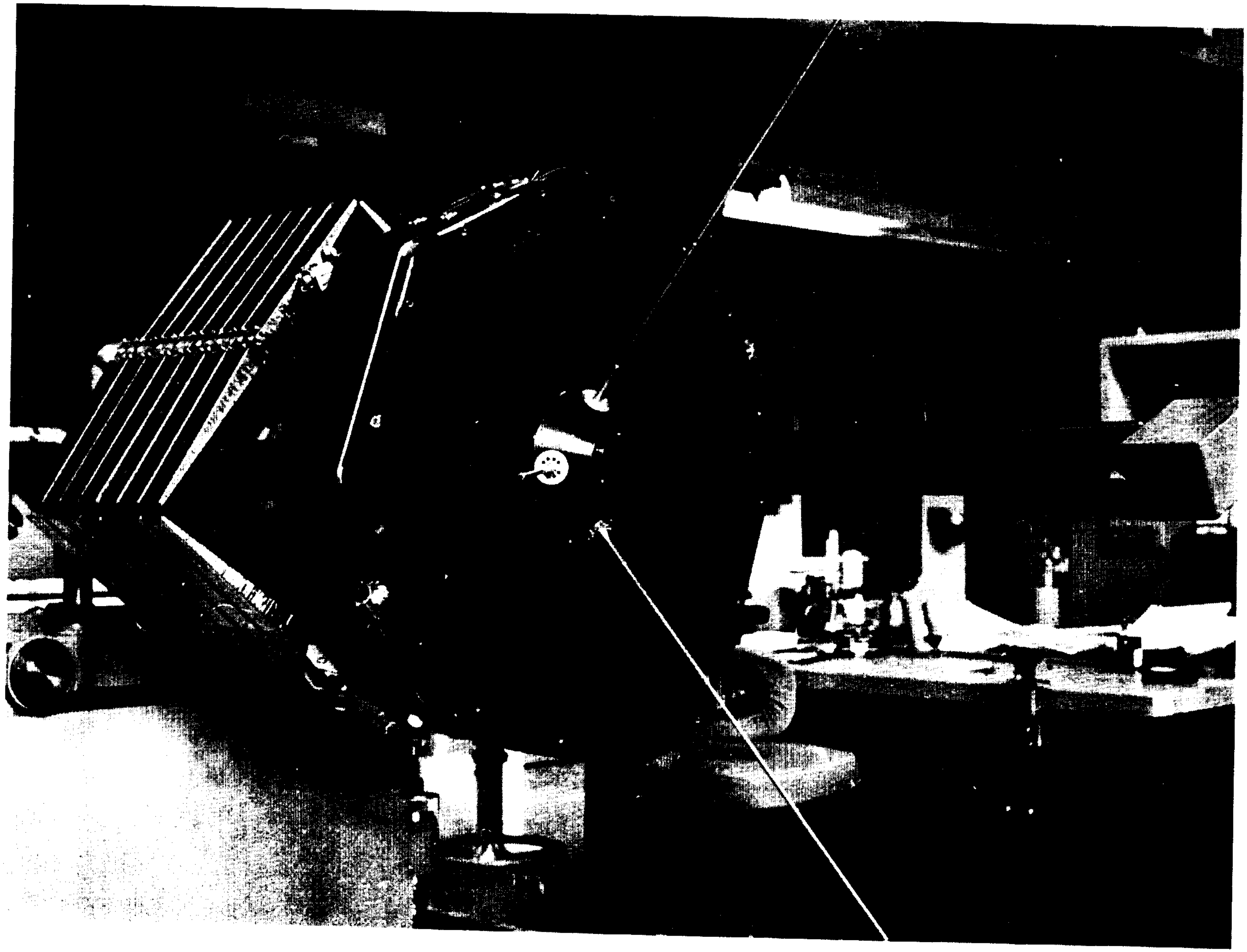
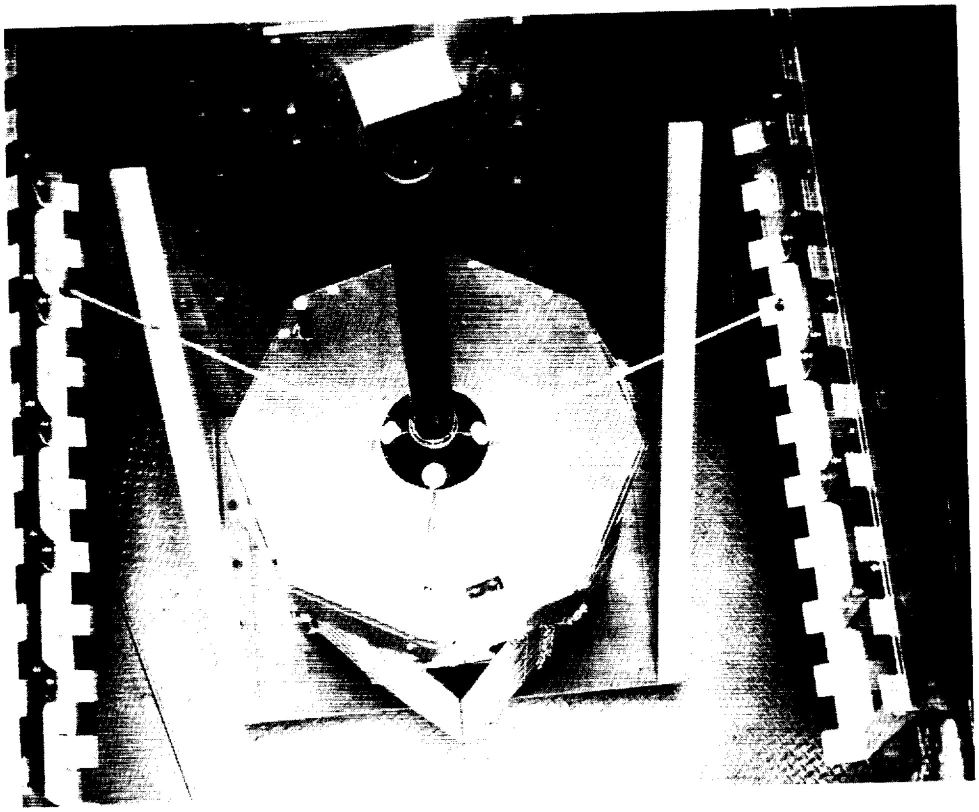
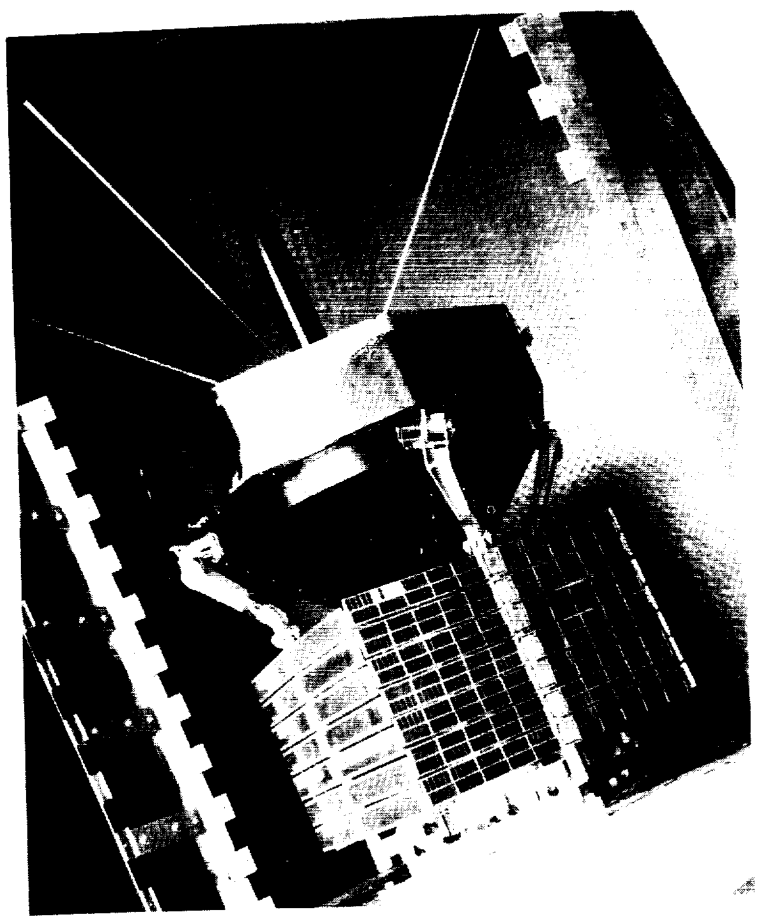


Figure 7. Typical Balancing Configuration
(S-3 Prototype Unit)



a. Mounting for Transverse Axes Acceleration



b. Mounting for Thrust Axis Acceleration

Figure 8. S-3 Prototype Unit Mounted on Centrifuge

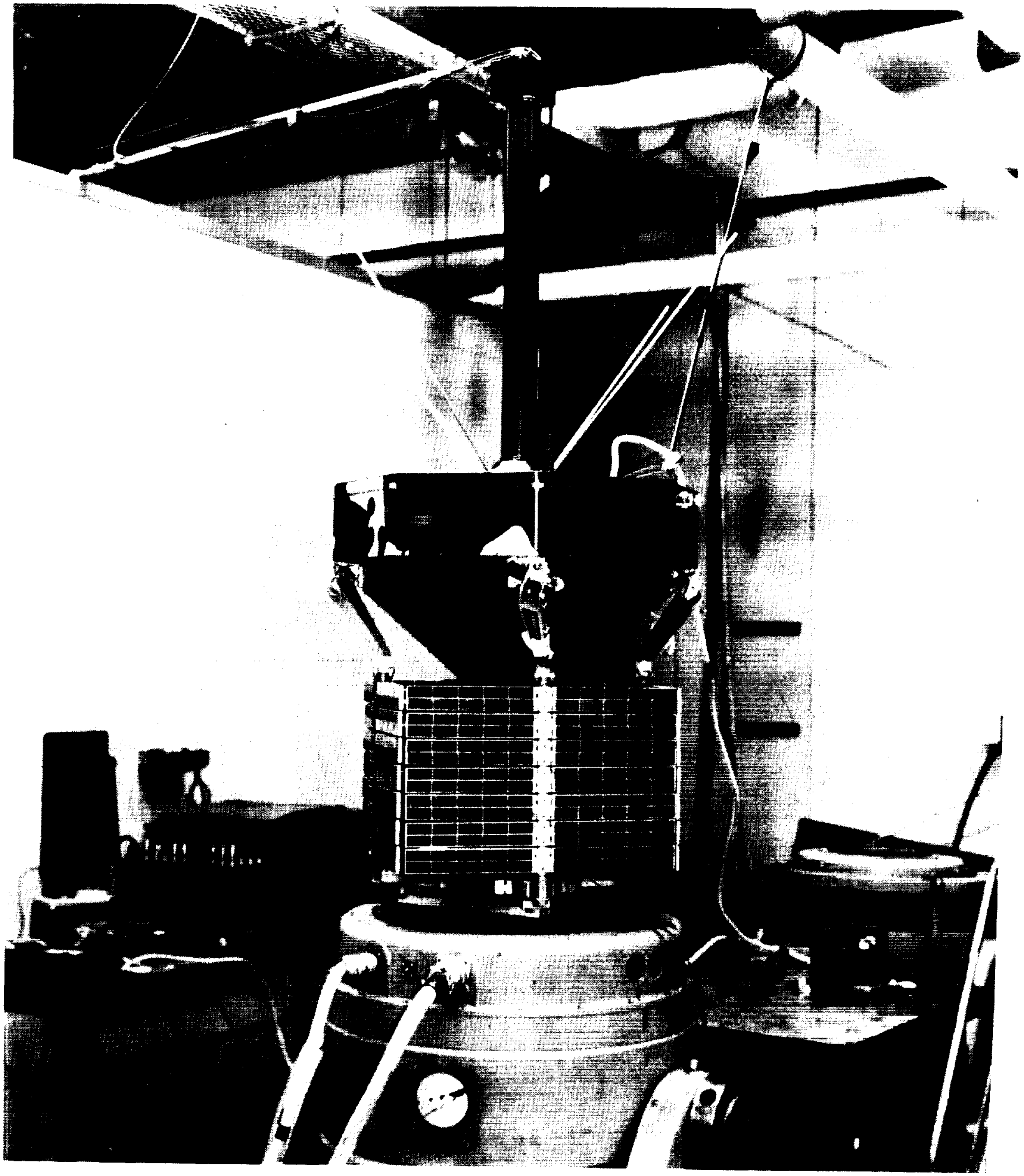


Figure 9. Thrust-Axis Vibration Configuration

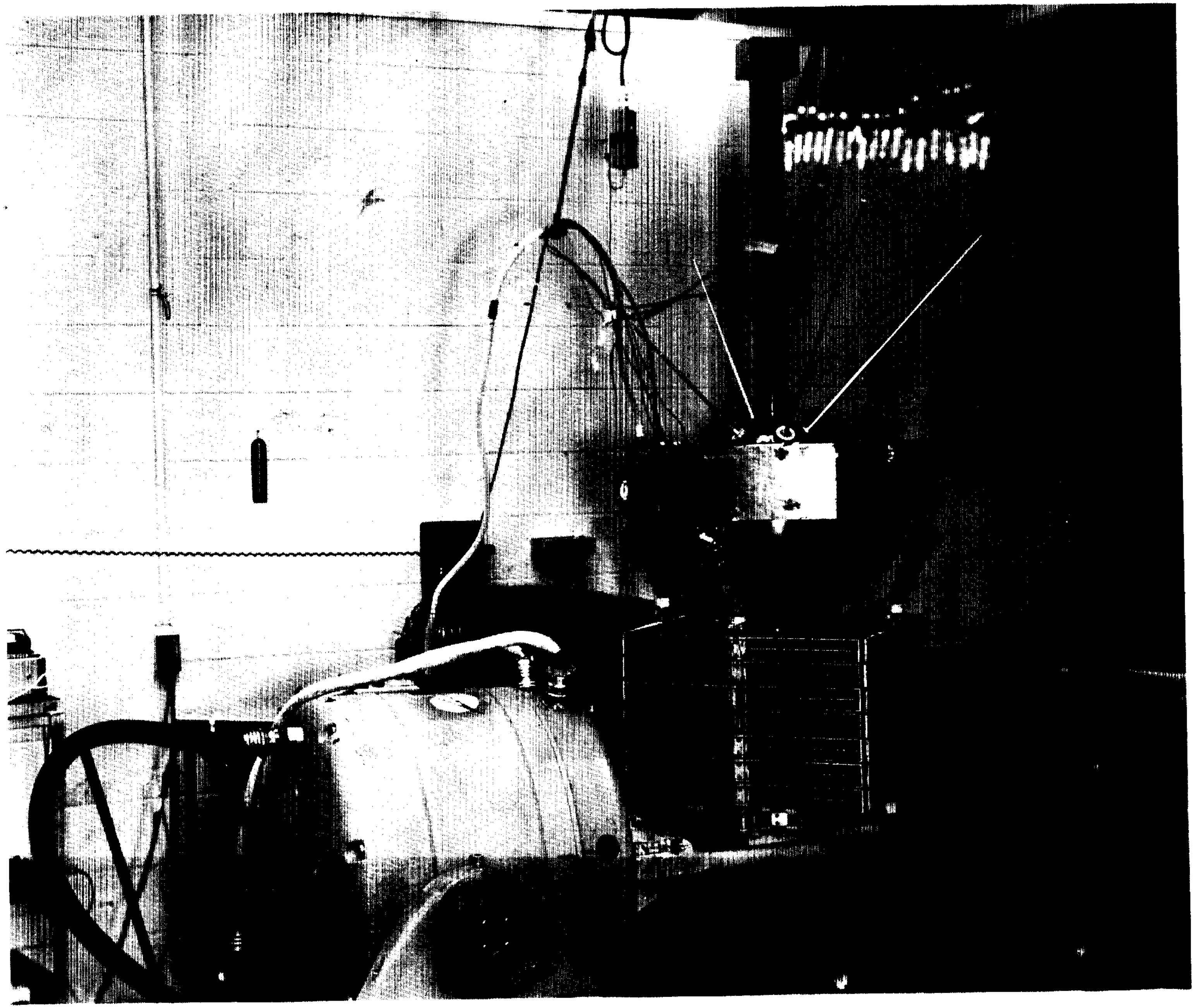


Figure 10. Typical Transverse-Axis Vibration Configuration

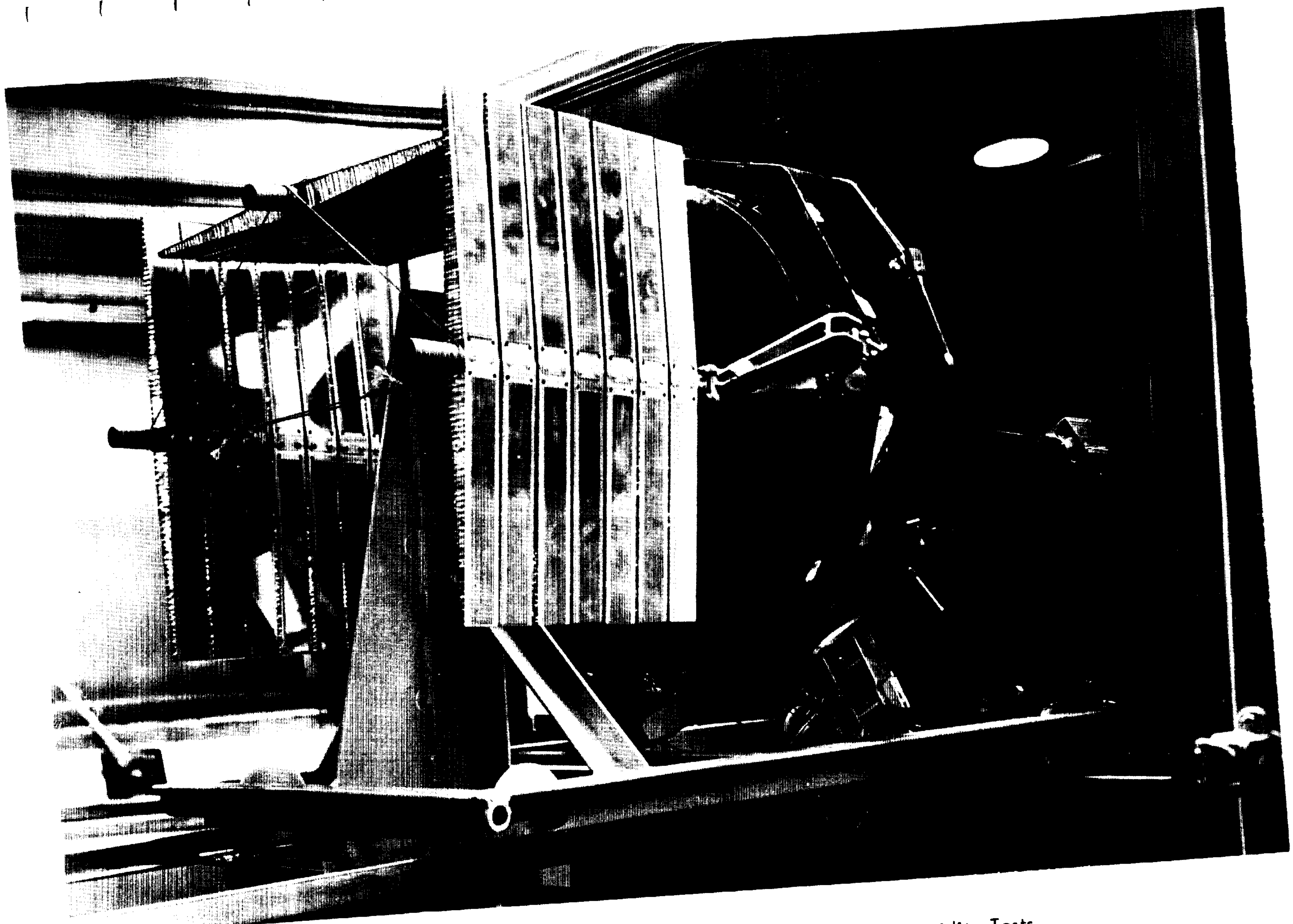


Figure 11. S-3 Prototype Unit Prepared for Temperature and Humidity Tests

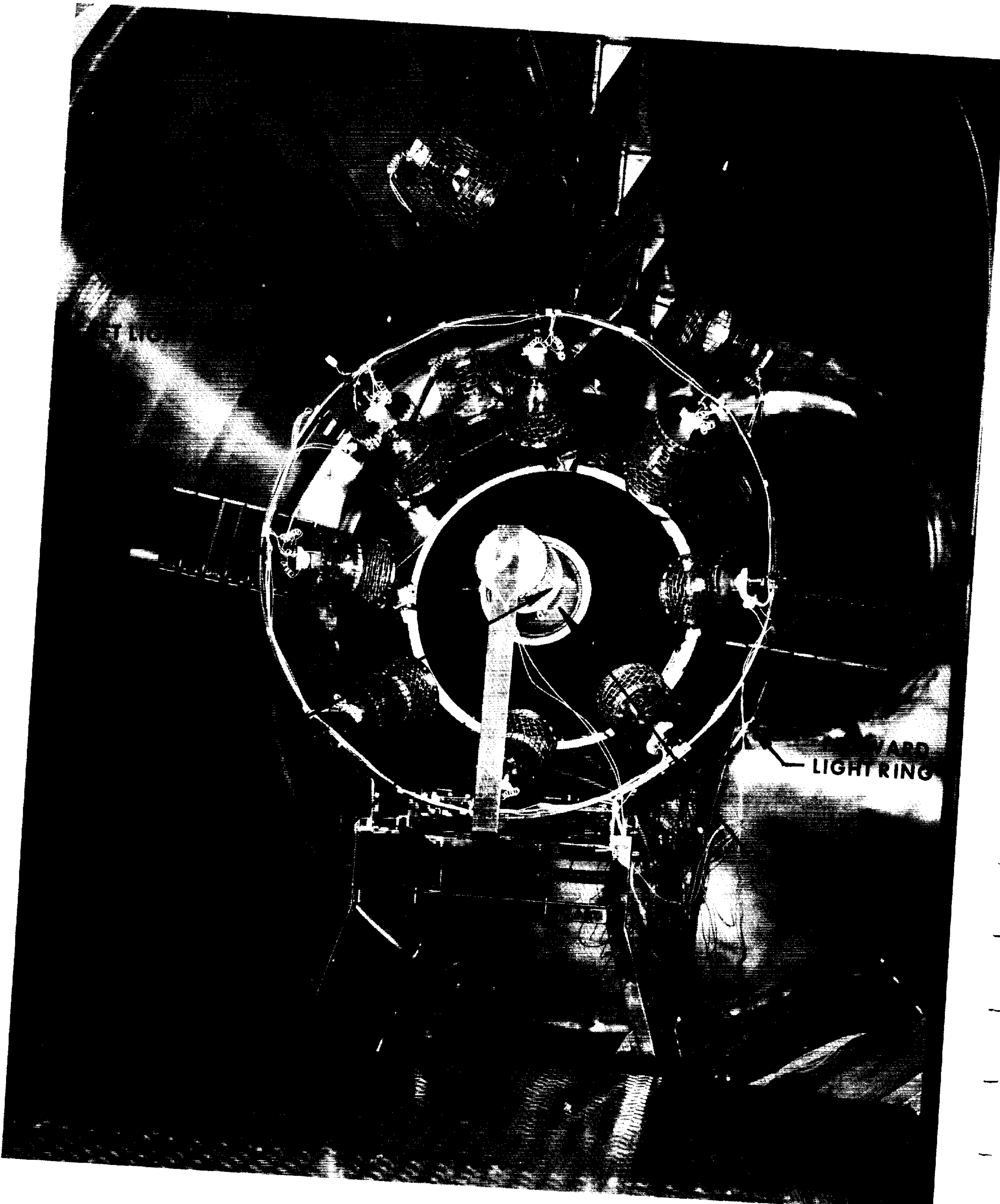


Figure 12. Typical Configuration for Thermal-Vacuum Tests
(S-3 Prototype Unit)

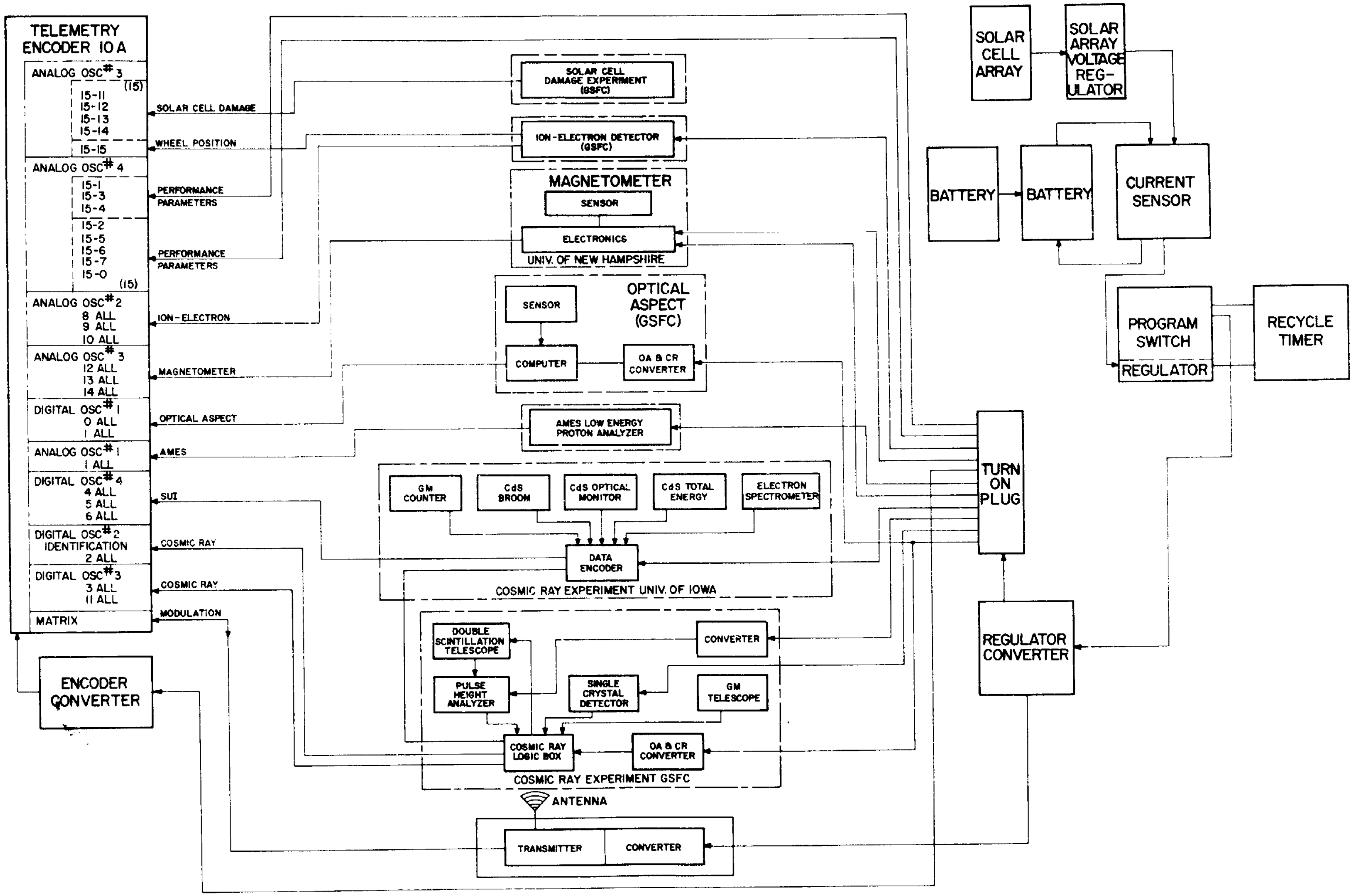
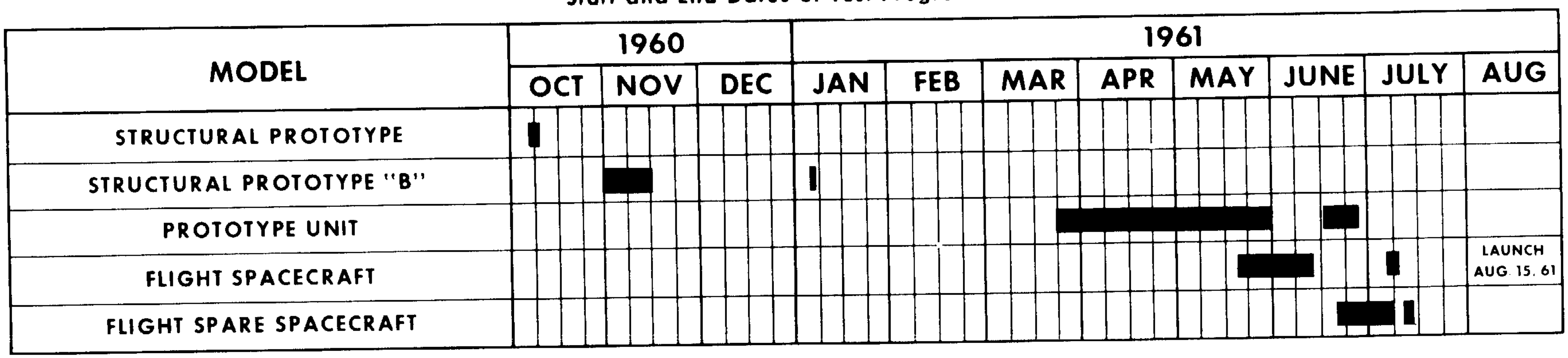


Figure 13. S-3 Energetic Particles Satellite System, Block Diagram (General)

Chart 1

S-3 ENVIRONMENTAL TEST PROGRAMS SCHEDULE

Start and End Dates of Test Programs



Start and End Dates of Individual Tests

MODEL	BALANCE			SPIN		ACCELERATION		VIBRATION			TEMPERATURE		THERMAL-VACUUM		SHOCK		HUMIDITY		RE-TEST		FINAL BALANCE	
	START	OFF	END'G	START	END'G	START	END'G	START	OFF	END'G	START	END'G	START	END'G	START	END'G	START	END'G	VIB	VAC	START	END'G
STRUCTURAL PROTOTYPE	6 OCT 60	--	7 OCT 60	--	--	--	--	8 OCT 60	--	8 OCT 60	--	--	--	--	--	--	--	--			--	--
STRUCTURAL PROTOTYPE "B"	1 NOV 60	--	3 NOV 60	6 JAN 60	6 JAN 60	15 NOV 60	15 NOV 60	3 NOV 60	5,6,7 NOV	9 NOV 60	--	--	--	--	--	--	--	--			--	--
PROTOTYPE UNIT	22 MAR 61	--	24 MAR 61	25 MAR 61	25 MAR 61	28 MAR 61	29 MAR 61	30 MAR 61	2 APR 61	3 APR 61	4 APR 61	6 APR 61	14 APR 61	24 MAY 61	31 MAY 61	31 MAY 61	16 JUNE 61	16 JUNE 61	24 JUNE 61	25 JUNE 61	--	--
FLIGHT SPACECRAFT	20 MAY 61	21 MAY 61	24 MAY 61	--	--	--	--	27 MAY 61	--	27 MAY 61	--	--	29 MAY 61	13 JUNE 61	--	--	--	--			7 JULY 61	8 JULY 61
FLIGHT SPARE SPACECRAFT	19 JUNE 61	--	21 JUNE 61	--	--	--	--	21 JUNE 61	--	23 JUNE 61	--	--	25 JUNE 61	7 JULY 61	--	--	--	--			11 JULY 61	13 JULY 61

Chart 2

SUMMARY OF THE FAILURES ENCOUNTERED DURING TESTING OF THE S-3 PROTOTYPE UNIT, FLIGHT SPACECRAFT & FLIGHT SPARE SPACECRAFT

		VIBRATION TESTS	THERMAL VACUUM TESTS					
			PRE-TEST CHECKOUT	CHAMBER EVACUATION	HOT (+35°C) TEST	COLD (-10°C) TEST	45° ASPECT TEST	135° ASPECT TEST
PROTOTYPE UNIT	TRANSMITTER	■				◨		
	PROGRAM SWITCH				◨			
	REGULATOR CONVERTER				◨			
	SOLAR ARRAY VOLTAGE REGULATOR				■			
	GM TELESCOPE	◢ ◢	◢					
	SINGLE CRYSTAL DETECTOR				◢			
	DOUBLE TELESCOPE						◢	
	PULSE HEIGHT ANALYZER				◢	◢		
	ION-ELECTRON DETECTOR	◢						
	MAGNETOMETER ELECTRONICS				◢			
	SUI OMNI COUNTER		◢	◢				
FLIGHT SPACECRAFT	DOUBLE TELESCOPE		◢					
	GM TELESCOPE			◢				
FLIGHT SPARE SPACECRAFT	TRANSMITTER						◨	
	BATTERIES/PROGRAM SWITCH						◨	
	ION ELECTRON DETECTOR							◢
	GM TELESCOPE	◢						

IF THE SUBASSEMBLY MALFUNCTION (OR OTHER UNSATISFACTORY PERFORMANCE) OCCURRED IN FLIGHT, IT COULD HAVE CAUSED:

■ A MISSION FAILURE

◨ A PARTIAL MISSION FAILURE BUT ONLY WITH AN IMPROBABLE SIMULTANEOUS COMBINATION OF CIRCUMSTANCES

◢ A LOSS OF DATA FROM THE AFFECTED EXPERIMENT

CHART DOES NOT INCLUDE MARGINAL AND/OR QUESTIONABLE DISCREPANCIES.

Chart 3
THERMAL-VACUUM TESTS
S-3 FLIGHT SPACECRAFT
Thermal Test Data and Available Flight Data

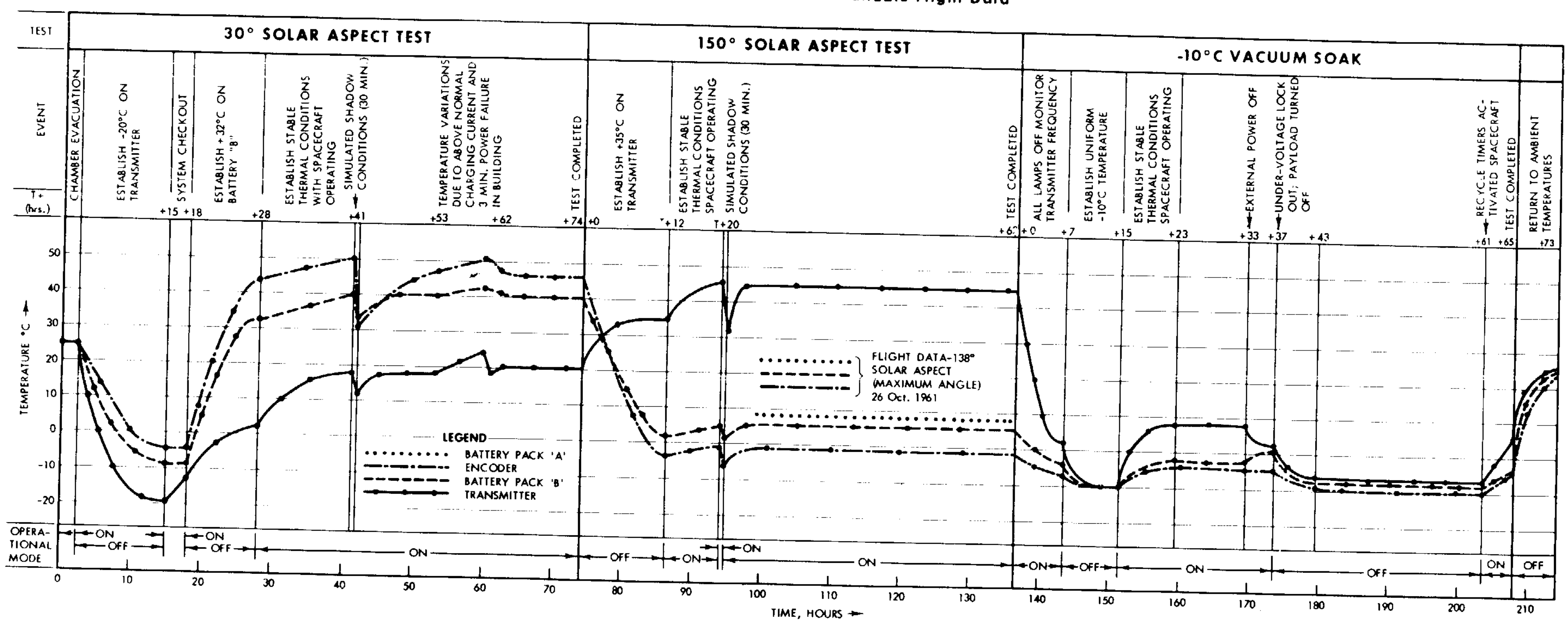


TABLE 1

Operational Temperature Extremes
Thermal-Vacuum Systems Tests

Location	Prototype Unit (°C)		Flight Units (°C)			
			Flight Spacecraft		Flight Spare Spacecraft	
	Minimum[1]	Maximum[2]	Minimum[3]	Maximum[4]	Minimum[3]	Maximum[4]
Telemetry Encoder	-2	+38	-4	+49.5	-4.5	+51.5
Regulator Converter	+13	+59	+9	+45	+5.5	+43
Battery "A"	-3.5	+38	+3	+36	-2	+44
Battery "B"	-2.5	+37.5	+2	+40.5	-4.5	+35
Pulse Height Analyzer	-2.5	+45	-3	+49.5	-4	+49
Transmitter	+6.5	+50.5	+5	+46.5	+4.5	+45

(1) During low temperature (-10°C) vacuum soak

(2) During high temperature (+35°C) vacuum soak except the Pulse Height Analyzer and Transmitter temperatures which occurred during the 45° Solar Aspect and the 135° Solar Aspect Exposures, respectively

(3) During low temperature (-10°C) vacuum soak

(4) During 30° Solar Aspect Exposure except the Transmitter temperature which occurred during the 150° Solar Aspect Exposure

APPENDIX A

PROJECT DESCRIPTION

ENERGETIC PARTICLES SATELLITE

S-3

REVISED

JULY 1, 1961

N-90,009

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D. C.

CONTENTS

ENERGETIC PARTICLES SATELLITE

S-3

1. TASK

   Energetic Particles Satellite S-3

2. PROJECT

   Energetic Particles

3. PROGRAM

   Geophysics

4. PROJECT MANAGER

   Paul Butler, Goddard Space Flight Center

   SCIENTIFIC ADVISOR

   Dr. Frank B. McDonald, Goddard Space Flight Center

   ASSISTANT PROJECT MANAGER

   Paul G. Marcotte, Goddard Space Flight Center

   PROJECT COORDINATOR

   Gerald W. Longanecker, Goddard Space Flight Center

5. SPACECRAFT (PAYLOAD) SYSTEMS GROUP

   Goddard Space Flight Center

6. PARTICIPANTS

   The working group, in addition to those listed in 4; consists of representatives from the following areas.

   6.1 <u>Instrumentation Agencies</u>

   Goddard Space Flight Center
   State University of Iowa
   University of New Hampshire
   Ames Research Center

Revised July 1, 1961

6.2 Goddard Space Flight Center

6.2.1 Spacecraft Technology Division

Systems Integration Branch
Flight RF Systems Branch
Flight Data Systems Branch
Thermal Systems Branch
Mechanical Systems Branch

6.2.2 Test and Evaluation Division

System Evaluation Branch

6.2.3 Tracking Systems Division

Systems Engineering Branch
Operations Control Branch

6.2.4 Data Systems Division

Data System Design Branch

6.2.5 Vehicle Liaison

7. OBJECTIVES

The primary objective of this satellite is to describe completely the trapped corpuscular radiation, solar particles, cosmic radiation, the solar winds, and to correlate particle phenomena with magnetic field observations. These objectives are further detailed as follows:

To map the particle intensity, including lower energy particles, in the radiation belts, in greater detail.

To study the time variations of the intensity of trapped radiation, and their relation to solar activity.

To determine the lifetime of particles in the trapped regions.

To look for evidence of local acceleration of charged particles.

To determine the frequency of occurrence of solar particle bombardments, especially at low intensity and low energy periods, which may not be recorded on ground monitors.

To study the possible injection of solar particles into the trapping regions.

To obtain data useful for studying the modulation mechanism of cosmic radiation.

To obtain data useful for evaluating the radiation effects on manned space flights.

8. DESCRIPTION

8.1 General

Satellite S-3 with its highly eccentric orbit extending from a perigee of 200 km (150 nautical miles) to a minimum apogee of 77,000 km (40,000 nautical miles) offers a unique opportunity to study the physics of fields and energetic particles in space. At apogee the spacecraft is essentially in interplanetary space beyond the earth's effective magnetic field. As the satellite moves away from apogee it passes through the trapped radiation regions; and at perigee it is located below this region.

The experiments in this spacecraft have beer carefully selected to cover the particle spectra from energies of a few ev to 10^9 ev. Simultaneous magnetic field measurements extending to a lower limit of several gamma will be made. In addition, the Ames Research Center is providing a solar plasma probe.

The repeated observations by this satellite of the solar wind, the interplanetary magnetic field, the distant regions of the earth's magnetic field and the particle population of interplanetary space and in the trapped radiation region should greatly enhance the understanding of these phenomena.

8.2 Experiments

8.2.1 Proton Analyzer   Ames Research Center Experiment (Dr. Michael Bader)

8.2.1.1 Objective

The purpose of this experiment is to measure proton flux and spectrum in space beyond 6 earth radii. The data obtained will increase our knowledge of proton concentrations in solar winds caused by solar flares. These data will be useful for correlating particle activity in space and in the Van Allen radiation belts with solar activity.

8.2.1.2 Principle of Operation

The proton concentrations as a function of kinetic energy are determined by admitting the protons through a slit of known dimensions in the satellite skin. A variable curved-plate electrostatic analyzer separates the particles according to their energy. This results in a particle current which is a function of the energy level and is measured by an electrometer circuit. By proper calibration of the analyzer in the laboratory, and given the geometrical and electrical characteristics, the particle concentration outside the spacecraft can be determined.

8.2.1.3 Range and Accuracy

There is at present a considerable uncertainty regarding spacecraft potential, especially in the radiatior belts; therefore, it is planned to maintain resolution down to 200 ev and provide an order of magnitude answer on concentrations at energies below this level. This effectively dictates a 20 Kev upper limit which is also the highest energy level expected from solar wind protons. A dynamic range of 10^4 is planned for the proton current measurement, which is sufficient to cover the extremes in expected solar proton fluxes. The basic accuracy of the current measurement is ±3 percent. Additional errors in computing flux may make this as great as ±5 percent. The energy measurement accuracy over the 0.2 to 20 Kev range is intended to be ±5 percent. The above accuracies are based on the assumption of 100 cps transmission bandwidth and a 1 rps spin rate. There will be no mass-analysis performed, but this experiment will be highly accurate for protons as these are believed to constitude at least 85 percent of the positive ion population. Since the flux at a given energy is inversely proportional to the square root of the mass, the error in using this figure for making a heavier particle correction should be quite small.

8.2.1.4 Weight, Size and Power Requirements

The analyzer package will be a rectangular box 3 x 4 x 2 inches with a total weight of 391 grams. The total power consumption is 145 milliwatts and will be supplied from +12 volt dc source with a regulation of ±1 percent.

8.2.2 Magnetometer University of New Hampshire Experiment (Dr. Laurence Cahill)

8.2.2.1 Objective

The purpose of this experiment is to measure the magnitude and direction of the earth's magnetic field between 3 and 10 earth radii as accurately as possible, to investigate the possible termination of the geomagnetic field in the vicinity of 10 earth radii, and to use these data in determining the existence of postulated extra-terrestial current systems and magnetic disturbances, particularly in relation to solar events and changes in particle intensities. These data will be used to accomplish the following objectives.

A study of the undisturbed magnetic field of the earth will be made to determine if the field is terminated by solar wind pressure within the range of measurement. The data will be studied for evidence of a "ring" current. Variations of such a current, both in spectral position and in time, would be investigated.

The data will be examined for rapid changes of the magnetic field in time. These might be interpreted together with information from ground magnetic observatories, as evidence for the propagation of hydromagnetic waves.

Time variations in the magnetic field will be compared with surface magnetic measurements and with records of solar activity in an attempt to discover possible correlations, particularly during magnetic storms. The time variations in the field will be compared with the variations in particle intensities for fixed locations between the two measurements.

The direction of the magnetic field will be available for comparison with directional characteristics of the particle intensities.

8.2.2.2 Principle of Operation

The magnetometer is a three-core device. Each of the three orthogonal sensors will produce an output voltage proportional to the magnitude of the component of the combined magnetic field along that sensor. The output voltages of the three sensors will each occupy a separate channel and will be combined after reception to form the total magnetic field vector.

8.2.2.3 Range and Accuracy

The range of measurements will be from a few gammas to 1000 gammas. The accuracy of the device is ± 10 gammas.

8.2.2.4 Weight, Size and Power Requirements

The magnetometer instrumentation will consist of two packages; a sensor package which is a rectangular box 3 x 3 x 4 1/2 inches, and an electronics package which is a rectangular box 3 x 4 x 6 1/4 inches. The sensor and the electronics package will weigh 640 grams and 760 grams respectively. The total power consumption, 405 milliwatts, is divided as follows:

260 mw, +12 VDC, 1 percent regulation;

80 mw, +12 VDC, unregulated.

8.2.3 Cosmic Ray Experiments State University of Iowa (Dr. Brian J. O'Brien)

8.2.3.1 Objective

The purpose of this experiment is to measure the characteristics of particle radiation over the entire spacecraft orbit. This radiation may be considered in three categories: Trapped Particles; Solar Particles; and Cosmic Rays. The characteristics of interest are the fluxes and energies of particles of various types, and a study will be made of their spatial and temperal dependence. Upon evaluation of the data, it may be possible to investigate the many geophysical problems and the many features of solar-terrestrial relationships.

8.2.3.2 Principle of Operation

The instrumentation presently consists of four geiger counters and three cadmium sulphide cells.

Sui GM Omnidirectional Counter. One geiger tube will have characteristics similar to one flown by the State University of Iowa on Explorer VII and by the University of Minnesota on Pioneer V. This will be regarded as a detector of particles arriving from every direction. It will measure protons above 20 Mev and electrons above 2 Mev.

Electron Spectrometer. One geiger will detect electrons between about 40 and 55 Kev which are focussed onto it by being deflected by the field of a small magnet. This tube will be heavily shielded, but another geiger will be housed alongside in identical shielding so as to monitor any very penetrating radiation background (protons above 50 Mev, electrons above 10 Mev). This tube will then be of interest in itself since it will measure cosmic rays and other very penetrating particles. The third geiger will measure electrons between 90 - 100 Kev.

Cadmium Sulphide Cells. The cadmium sulphide cells are small crystals whose electrical conductivity increases as they are bombarded with ionizing particles. The conductivity is measured by applying a steady voltage to the crystal. Charge flowing through the crystal builds up on a condenser until a critical voltage is reached and the condenser discharges through a glow tube. The rate of discharge thus increases with the conductivity and hence the ionization energy lost in the crystal also increases the rate of discharge. The crystals are uncovered and hence can detect particles to very low energies, e.g., electrons and protons down to energies of the order of 100 ev or less. See Table 1.

CdS Total Energy. One cadmium sulphide cell is used to measure the total energy flux of both protons and electrons incident on it.

CdS Broom. Another cell has a magnet which deflects electrons below several hundred Kev from striking the crystal. This is then essentially a low-energy proton detector.

CdS Optical Monitor (alteration or correction). A third cadmium sulphide cell looks in the same narrow region of the sky as do the other two. This is fitted with a transparent shield so that the corrections for the effects of light (e.g., earthlight or sunlight) striking the other two photosensitive cells may be made accurately.

Sui Encoder. The spectrometer and the cadmium sulphide cells view narrow positions of the sky perpendicular to the axis of spin of the satellite. The detectors will be selected, two at a time, to feed two scaling units for 10.24 seconds. The scalers will each be read out twice and the information telemetered as a sequence of binary bits. The scalers will be reset to zero, the next pair of detectors selected. This information is fed to an appropriate digital subcarrier oscillator for transmission on telemetry channels 4, 5, and 6.

8.2.3.3 Range and Accuracy

The energies of particles of different types which can be detected by the State University of Iowa apparatus are tabulated in Table 1.

Table 1. Particle Energies Detected

Detector	Particles Detected	Detector Response (Approximately)
CdS Total Energy	Electrons Protons	above about 100 ev above about 100 ev
CdS Broom	Protons Electrons	above about 100 ev above 500 Kev
G.M. Tube (triple) or Electron Spectrometer	Electrons Electrons Protons Electrons	a) 40 Kev $<$ E $<$ 55 Kev b) 90 Kev $<$ E $<$ 100 Kev c) above about 50 Mev above about 10 Mev
G.M. Omnidirectional Counter	Protons Electrons	above about 20 Mev above about 2 Mev

The storage capacity of the apparatus is for 2^{18} counts in 10.24 seconds, i.e., a maximum counting rate of about 25,000 counts/seconds. The selectivity of the geiger counters is such that, on the basis of present data, it is expected that each will be able to measure intensities from the cosmic ray level right up to the most intense fluxes in the radiation zones.

The lowest counting rate of the CdS detectors is set by their dark current. This will be measured on the present scheme. The dynamic range of the detectors is about 10,000 and the apertures are chosen so that they can measure adequately up to the estimated maximum energy flux in the most intense regions of the radiation zones without damaging the cells.

The accuracy of each measurement depends upon the counting rates and on the extent to which samples at one-minute intervals can be treated as identical. In the radiation zones, an accuracy of several percent should be common. In all cases the error in any one measurement will be extremely small in comparison with the dynamic range covered by each detector (of the order 10,000 to 1).

8.2.3.4 Weight, Size and Power Requirements

The weights for the State University of Iowa experiment are listed below:

CdS Optical Monitor CdS Total Energy Detector CdS Magnetic Broom	494 grams
Electron Spectrometer	743 grams
G.M. Counter (Omnidirectional)	292 grams
Encoder plus hardware	1540 grams

The State University of Iowa apparatus is housed in six modules, comprised of five detector modules and one encoder package. The power requirements are 258 milliwatts, at +6.5 v ±5 percent.

8.2.4 Cosmic Ray Experiments Goddard Space Flight Center (Dr. Frank B. McDonald)

8.2.4.1 Objective

At the present time it is felt that the most important problems in cosmic rays are the nature of the accelerating mechanism and the nature of the modulation mechanism which produces the 11-year variation and the Forbush type decrease. An accelerating mechanism which can produce particles with energies up to 10^{18} ev and a modulation mechanism which can influence particles with energies greater than 10^{10} ev would both appear to have important astrophysical implications. The theoretical explanations for these phenomena are hopelessly inadequate at this time and it is clear that additional experimental information is needed.

The sun seems to be a very important source of low energy cosmic rays and the cosmic ray modulation mechanism appears to be intimately connected to solar events. If one understands the mechanism by which solar cosmic rays are produced and the connection between solar activity and cosmic ray intensity changes, then one probably has the necessary clues to understand the general origin of cosmic rays.

It is felt that with an effective cosmic ray monitoring program established beyond the effects of the earth's magnetic field, both the production and modulation of cosmic rays can be studied with the same set of experiments. In each case the parameters that should be measured are the charge spectra and the energy spectra of the cosmic radiation as a function of distance from the earth, time and direction. There are also indications that many of the effects of solar related phenomena are transmitted via the emission of a solar plasma or a solar wind and it is of the greatest importance to make simultaneous magnetic field and plasma measurements. The apogee portion of the proposed S-3 trajectory will be used to carry out these objectives.

8.2.4.2 Principle of Operation

Double Telescope

The cosmic ray package will consist of three basic detection units; the Double Telescope, a Single Crystal Detector, and a GM Telescope. The first detector is a double scintillation-counter telescope in which the pulse from one of two counters is selected for a given event. This unit provides the following information: the total cosmic ray flux; the flux of fast protons with energies greater than 700 Mev; the proton differential energy spectrum in the region 70-750 Mev; and the low energy portion of the Alpha particle differential energy spectrum. See Table 2.

When a particle traverses the two scintillators a coincidence is formed and the pulse height from one of the scintillation counters is processed by the GSFC 32 channel analyzer. Data is accumulated in the analyzer's magnetic core memory for a 4-minute period and is then read out serially. Read-out is nondestructive. Channel capacity is 2^{16} per channel.

Single Crystal Detector. In order to extend the proton energy spectra data down to 1 Mev, a thin CS-I scintillation counter is used. The pulse height distribution of the incident particles is obtained in the region 100 Kev to 20 Mev by means of a sliding channel pulse height analyzer. This unit will also provide information on low energy solar gamma rays.

This detector, or scintillation counter, is connected to an integral discriminator whose bias is furnished by an eight-level staircase generator. A data accumulator (cosmic ray logic box) is subcommutated between the eight levels and two Geiger counter inputs of the GM telescope. In each case the actual number of counts per unit time is transmitted. Appropriate identification is also provided for each readout. When one input of the multichannel analyzer is read-out, the other inputs are disconnected.

GM Telescope. The third unit consists of two Anton 1003 Pancake type geiger counters. One of these is shielded with 2 grams/cm^2 of appropriate material. The effective geometric factors of these counters are several orders of magnitude larger than those in the State University of Iowa package and are intended to be cosmic ray monitors. The rate of the shielded, and the coincidence rate of the two counters will be telemetered. These units furnish a check on the information received from the scintillation counter units.

8.2.4.3 Weight, Size and Power Requirements

Table 2 gives the weight, size, and power requirements along with the objective and range of the detector.

Table 2

Experiment	Experimental Objective	Weight (grams)	Size (inches)	Power (watts)
Double Scintillation Telescope	Measures total cosmic ray flux	765	20x2.5D	1.25
	Measures proton energy spectrum in 70 to 750 Mev			
	Measures low-energy spectra, alpha particles			
	Measures total flux fast protons in region above 700 Mev			
Single Crystal Detector	Proton and electron energy spectrum 100 Kev $< E <$ 20 Mev	615	4x2.5D	.200
	Low energy gamma ray			
GM Telescope		400	5x2.5D	.300
shielded	Proton flux $>$ 75 Mev Electron $>$ 8 Mev			
coincidence	Cosmic ray flux $>$ 75 Mev			

8.2.5 Optical Aspect Goddard Space Flight Center (Mr. James S. Albus, Mr. David H. Schaefer)

8.2.5.1 Objective

The purpose of the optical aspect system is to determine the orientation in space of the spacecraft as a function of time.

8.2.5.2 Principle of Operation

The orientation of the spacecraft will be determined by using solar sensors only. Six photodiodes are to give 180 degrees digital indication of the sun's elevation with respect to the spin axis of the satellite - the 180 degrees from pole-to-pole being divided into 63 parts. The time within the telemetry frame of the sun's appearance is also coded in binary form. Read-out of all the time and position information will be on two telemetry channels.

The system consists of two basic parts. The first is a digital solar aspect sensor, consisting of a light mask and a number of photo diodes placed behind the light mask so that each photo diode sees only the portion of the light mask directly behind it.

The second part is a digital computer having memory and logic for determining the time at which a photo diode sees the sun and for remembering which photo diode had the input.

The "time" or period data is read-out from telemetry channels 0 and 1; channel 0 holding the data, channel 1 being blank. In the succeeding telemetry frame, channels 0 and 1 both hold the sensor "position" data.

8.2.5.3 Accuracy

The overall accuracy will be better than 5 degrees in azimuth and elevation.

8.2.5.4 Weight, Size and Power Requirements

The sensor package is basically rectangular 2 x 2.5 x 2 inches, weighing 248 grams; the electronics is a printed card, size 5 x 7 inches, and is integral with the Telemetry Encoder and Converter electronics. The maximum power requirement is 20 milliwatts. (average is 2-3 milliwats)

8.2.6 Ion Electron Detector Goddard Space Flight Center (Leo R. Davis)

8.2.6.1 Objective

The objective is to measure particle fluxes, types and energy as a function of direction, time and position below, in, and above the Van Allen radiation belts. This detector is most sensitive to the low energy particles which have not been directly measured to date and yet have been indicated to be in the inner and outer radiation belts.

8.2.6.2 Principle of Operation

The ion-electron scintillation detector consists of a powder phosphor, ZnS(Ag), settled on an RCA 6199 photo-multiplier tube which is located behind a stepping absorber wheel. The dc current and pulse counting rates are measured simultaneously for each absorber position.

Ion counting rates for two trigger levels are registered for seven absorber thicknesses from which ion types and energy spectra can be deduced. In these measurements electrons are discriminated against by the phosphor thinness ($5 mg/cm^2$) and the phosphor characteristic of the emitted light decay time being inversely proportional to the square of the ionization density.

The electron energy flux is obtained by scattering the incident electrons off a gold plate (ions will be absorbed) into the phosphored photo-multiplier tube from which dc currents are measured. Electron energy spectra can be deduced by comparing the responses from six absorber thicknesses.

The total energy flux is obtained for seven absorber thicknesses by measuring the photo-multiplier dc current.

8.2.6.3 Range and Accuracy

The ion detector is operative over the energy range of 100 Kev to 1 Mev for protons with maximum counting rates of 10^5 cps in each channel.

The electron detector with a dynamic range of 10^5 is operative for electrons between 10 Kev and 100 Kev. For average photo-multiplier voltage, the minimum detectable energy is 10^{-2} ergs/seconds.

The total energy flux detector with a dynamic range of 10^5 is operative over the energy range of 30 Kev to 1 Mev for protons, and 10 Kev to 100 Kev for electrons. For average values of photo-multiplier voltage, the minimum detectable flux is 2×10^{-4} ergs/seconds. The detectors and their range of response is shown in Table 3.

Table 3

Exp. Objective	Detected Particles	Detector Response
Ion Detector	Protons	100 Kev $<$ E $<$ 1 Mev
Electron Detector	Electrons	10 Kev $<$ E $<$ 100 Kev
Total Energy Detector	Protons	30 Kev $<$ E $<$ 1 Mev
	Electrons	10 Kev $<$ E $<$ 100 Kev

8.2.6.4 Weight, Size and Power Requirements

The detector has a total weight of 1362 grams and is housed in a 5-3/8 x 6-7/8 x 3-1/8 inch container. Power requirements are 28 ma, average, from the main battery pack, at 12.6 to 20 volts. The detector is a constant current load.

8.2.7 Solar Cell Experiment Goddard Space Flight Center (Mr. G. W. Longanecker)

8.2.7.1 Objective

The purpose of this experiment is to measure the effects of the the deterioration of solar cells caused by direct exposure to the radiation in the Van Allen belts.

8.2.7.2 Principle of Operation

The experiment consists of four strips of silicon solar cells, with ten cells per strip, mounted on the surface of the spacecraft (Figure 1). One strip of ten cells is unprotected while the remaining three strips are protected by 3, 20 and 60 mil thick glass respectively. During the life of the spacecraft it will be possible to compare the effectiveness of the glass filters in preventing degradation of the solar cells due to radiation.

The telemetry has provision for four voltage measurements on a time sharing basis.

8.3 Spacecraft Structure Goddard Space Flight Center (Mr. F. T. Martin)

8.3.1 External Configuration and Dimensions

A preliminary layout of the spacecraft is illustrated in Figures 1 and 2. An octagon walled platform, fabricated from nylon honeycomb and fiberglas, houses most of the instruments and electronics. They are mounted on the periphery of the platform in order to obtain the highest possible roll moment of inertia and assure spin stability about the roll axis.

The transmitter is located in the base of the spacecraft. Thus, heat generated by the transmitter is dissipated through the structure and aluminum lower cover of the spacecraft.

A magnetometer package, containing three orthogonally mounted saturable core magnetometers and calibration coils, is located forward of the platform on a boom to reduce field effects from the electronics and instruments.

Four spring loaded solar celled paddles extend from the main structure. The paddles are oriented to allow a uniform solar cell projection area at any spacecraft-solar attitude. The paddles are folded along the last stage rocket to permit their installation within the nose fairing (Figure 3.) They are erected during flight. Attached to the side of the one piece upper aluminum honeycomb cover is the despin device which will reduce the roll rate to approximately 18 rpm after last-stage burnout.

8.3.2 Static and Dynamic Characteristics

Weight	83 lbs.
C.G. (P.E.)	12-1/8 in.
C.G. (P.F.)	10-9/16 in.
I_{roll}(P.E.)	3.49 slug-ft.^{2}
I_{roll}(P.F.)	1.87 slug-ft.^{2}
I_{1-3} (P.E.)	2.47 slug-ft.^{2}
I_{1-3} (P.E.)	2.26 slug-ft.^{2}
I_{2-4} (P.E.)	2.70 slug-ft.^{2}
I_{2-4} (P.F.)	2.34 slug-ft.^{2}

NOTES:

Weight includes balance weights

P.E. = Paddles extended

P.F. = Paddles folded

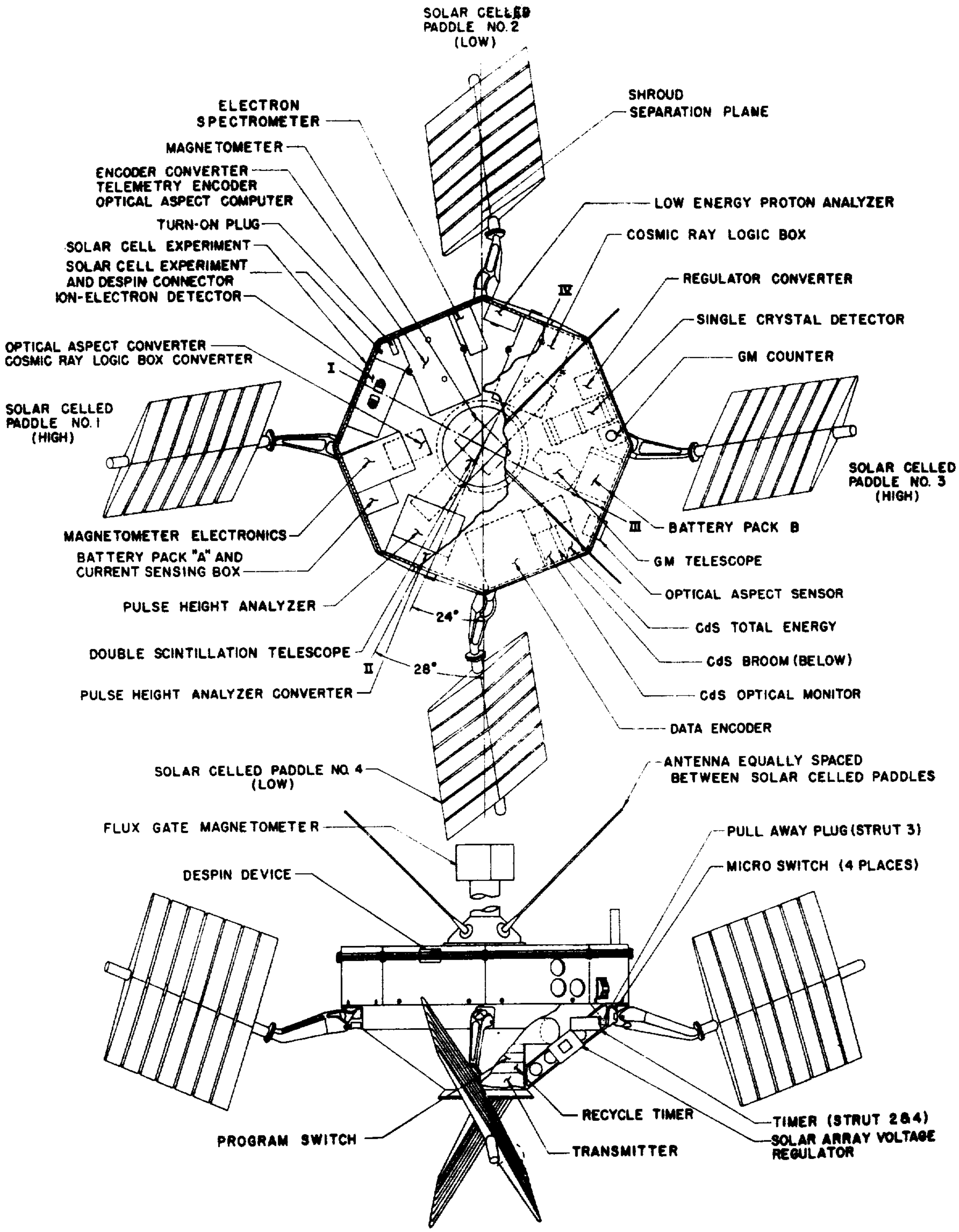


Figure 1 - S-3 energetic particles satellite

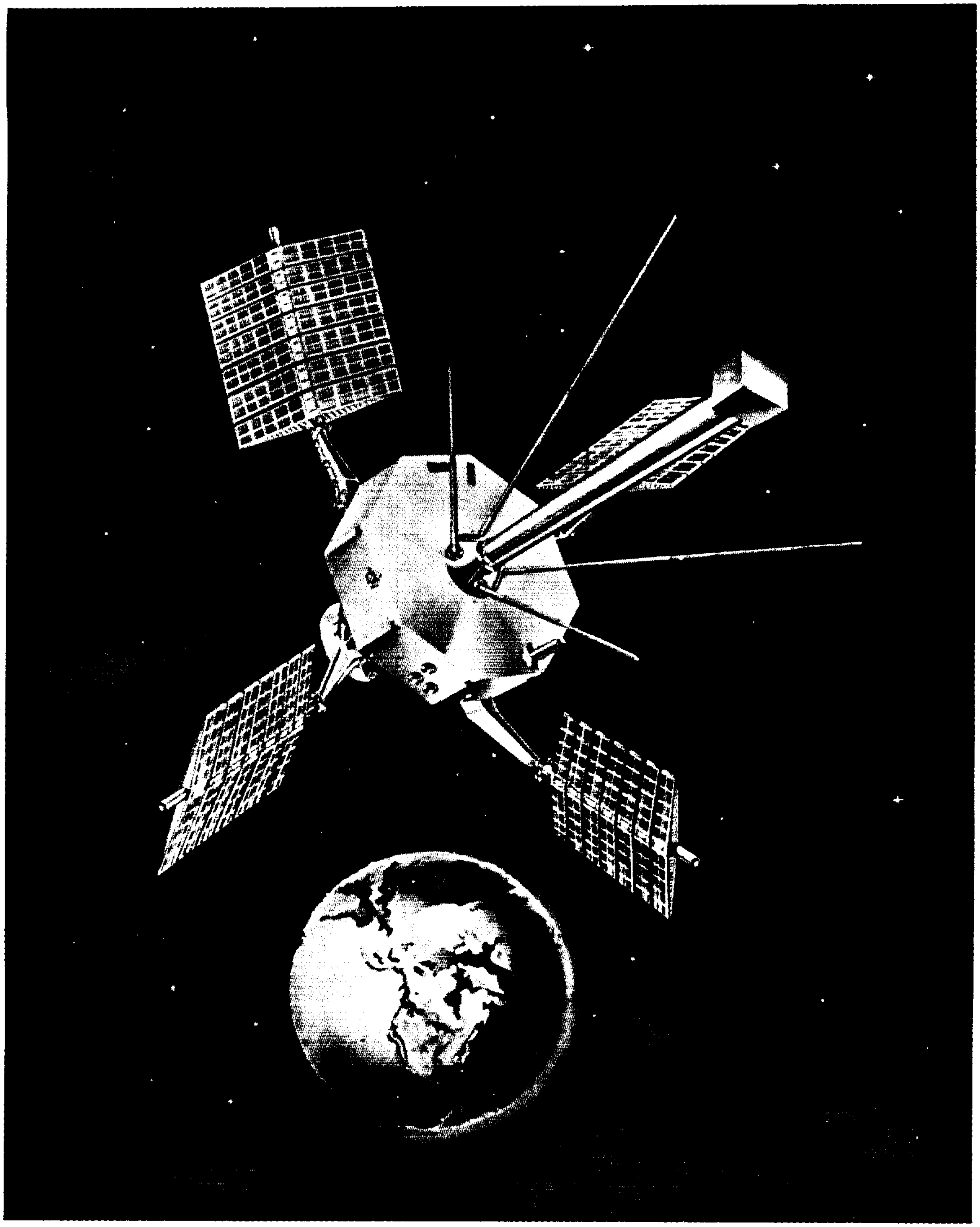


Figure 2 - Artists conception of the S-3 satellite

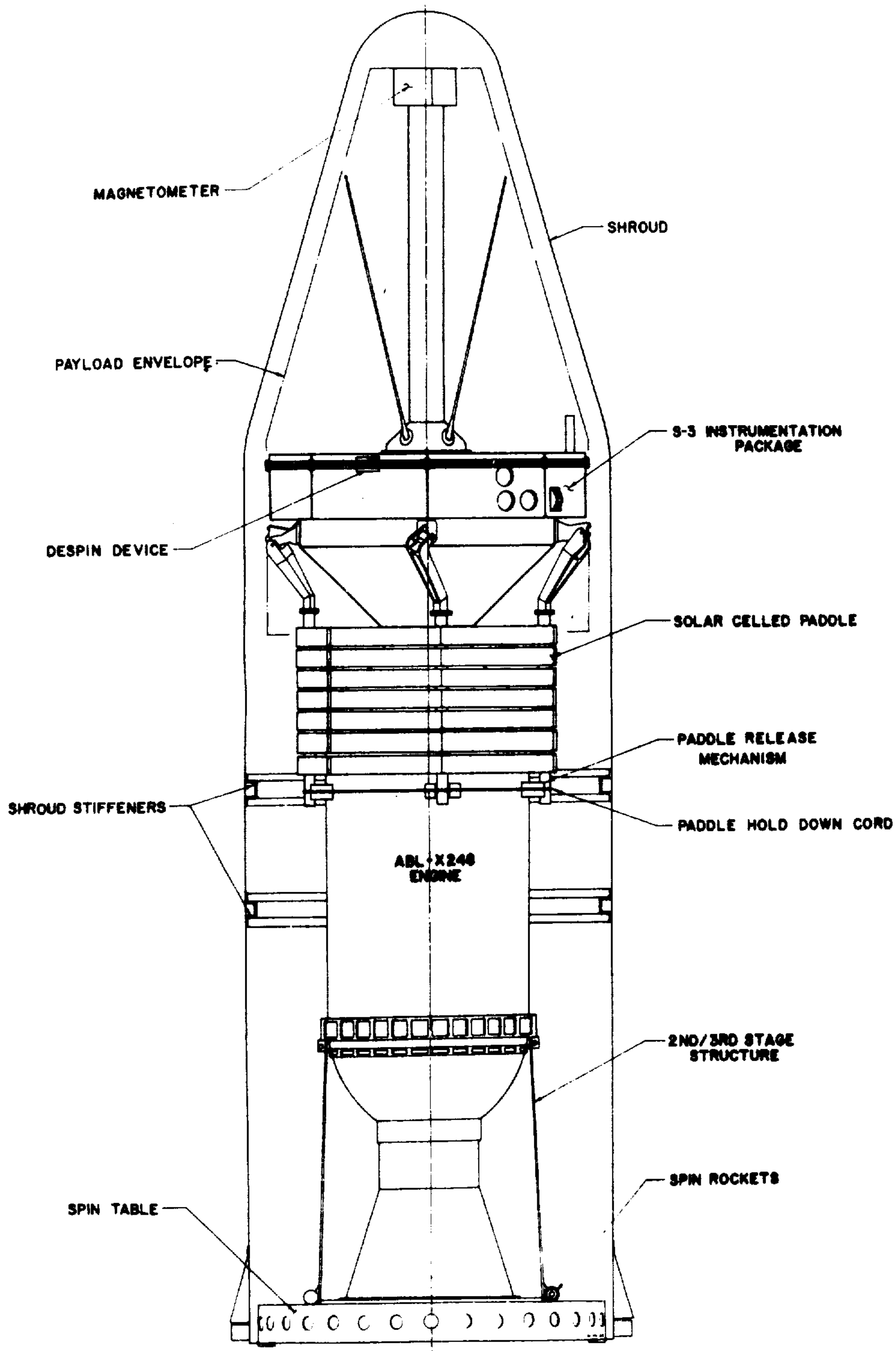


Figure 3 - S-3 spacecraft installation

1-3 = High paddle axis

2-4 = Low paddle axis

C.G.: From separation plane

8.3.3 Thermal Characteristics of Spacecraft Surfaces

The surface thermal characteristics are variable and are divided into A/e ratios for top, bottom, and sides. Effective A/e = 1.17. Evaporated aluminum, aluminum paint, and white paint are used to coat the external surfaces in a special pattern for thermal control.

8.4 Telemetry Goddard Space Flight Center (J. E. Scobey)

The telemetry system on Satellite S-3 operates continuously so that all data transmission is in real time. The system is of the pulse-frequency-modulation (PFM) time-division multiplex type, meaning that the modulation is composed of bursts of frequency separated in time by periods of no oscillation. Equal time intervals are devoted to the duration of a burst and to a period of no oscillation except for a synchronization reference composed of a 50 percent shorter period of no oscillation followed by a 150 percent longer burst. This reference defines the origin of each frame. A frame is defined as 16 sequential bursts, each burst representing a channel.

The complete telemetry encoding format encompasses 16 frames of 16 channels per frame. The bursts are 10 milliseconds in duration and the burst frequency range is 5 to 15 kc. The intelligence is in the burst frequency. Although the maximum sampling rate afforded by this encoding system is about 50 samples per second, the the S-3 experiment channels for the most part are not supercommutated; the sampling rate is once per frame or about 3 per second. An exception is the Single Crystal and G-M Counter of the Double Telescope Experiment which is supercommutated to a sampling rate of twice per frame. For the so-called "housekeeping" performance variables of the satellite for which only a low sampling rate is required, one channel is subcommutated by 16; hence the 16 frames to complete an encoding format (Figure 4). A second level of encoding is accomplished for some experiments within the signal conditioning electronics before application to the telemetry encoder. This processing is not considered a telemetry encoding operation but rather the programming of an appropriate experiment to a fixed routine, hence will not be discussed here.

The pulse frequency generators of the telemetry encoder are of two types - analog and digital. Both types are oscillators utilizing square-loop magnetic cores and transistors as active elements. Economy of power drain is realized by activating only one oscillator at a time from appropriate gating waveforms. The oscillator complement of the telemetry encoder is 5 analog and 4 digital oscillators. The overall bit rate of the encoded data format is about 300 bits per second. Accuracy of the analog oscillator is about ±1 percent or about one part in 2^7; of the digital oscillator, one part in 2^3. With three discrete binary inputs, the 8-level digital oscillator naturally lends itself to the encoding of information which occurs in digital form, such as three binary scalers of an experiment which record counts or the number of events, as a

FRAME

	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1	0	CHANNEL
DIGIT	OPTICAL ASPECT																0
DIGIT	OPTICAL ASPECT																1
DIGIT	IDENT (DOUBLE TEL-SINGLE XTAL & GM TELE)																2
DIGIT	DOUBLE TELESCOPE-SINGLE XTAL & GM TELE																3
DIGIT	SUI																4
DIGIT	SUI (ALL SUI EXPERIMENTS SUB-COMMUTATED ON THESE CHANNELS)																5
DIGIT	SUI																6
ANAL	AMES (PROTON ANALYZER)																7
ANAL	I & E (TOTAL ENERGY)																8
ANAL	I & E (1 MEV-ION)																9
ANAL	I & E (100 KEV-ELECTRON)																10
DIGIT	DOUBLE TELESCOPE-SINGLE XTAL & GM TELE																11
ANAL	MAGNETOMETER																12
ANAL	MAGNETOMETER																13
ANAL	MAGNETOMETER																14
ANAL	PP_{15}	PP_{14}	PP_{13}	PP_{12}	PP_{11}	PP_{10}	PP_9	PP_8	PP_7	PP_6	PP_5	PP_4	PP_3	PP_2	PP_1	PP_0	15

CHANNEL 15 PP_0 - PP_{15}

PP_{15}	TEMP. BATT. COMPART. NO. 1
PP_{14}	TEMP. BATT. COMPART. NO. 2
PP_{13}	TEMP. ANALOG OSC.
PP_{12}	TEMP. SOLAR ARRAY
PP_{11}	VOLT. REG. (2KC) 26.2V
PP_{10}	AMPS. BATT. DISCH.
PP_9	AMPS. SOLAR ARRAY 0.8A
PP_8	VOLT. REG. SUPPLY (-17.8V)
PP_7	POSITION I & E WHEEL
PP_6	SOLAR PATCH AMPS. (NO. 4)
PP_5	SOLAR PATCH AMPS. (NO. 3)
PP_4	SOLAR PATCH AMPS. (NO. 2)
PP_3	SOLAR PATCH AMPS. (NO. 1)
PP_2	VOLT. REG. SUPPLY (+6.5V)
PP_1	VOLT. REG. SUPPLY (+12V)
PP_0	VOLT. BATT. TERM.

Figure 4 - Telemetry channel assignments

G-M scaler. On the analog oscillators, selective gating of the frequency control input voltage (0 to +5 volts) of the oscillators is used to subcommutate several channels onto one oscillator. Examples of subcommutated channels are the three Ion and Electron Spectrometer outputs and the three Magnetometer outputs. Subcommutation is also done on 16 channels of "housekeeping" performance variables, 8 each on one of two oscillators.

Designed to operate in a temperature environment of from 0° to +50° C, the power drain of the encoder and converter is approximately 270 milliwatts. Weight of the encapsulated three decks of encoder electronics plus a fourth deck composed of the encoder power converter is approximately 3 pounds.

The output of the telemetry encoder provides time-division multiplex bursts of oscillation to phase modulate the transmitter to an index of ±57 degrees peak-to-peak. Total radiated power is 2 watts at a carrier frequency of approximately 136 Mc. Synchronous detection is used to receive the signal at a telemetry receiving station where the translated signal is recorded on magnetic tape.

At the decoding and data reduction ground facility a bank of contiguous comb filters, each of 100 cps bandwidth, is used in conjunction with an auction circuit such that the effective noise bandwidth is reduced to a fraction of the information bandwidth. Linear detection is employed ahead of the filter bank to maintain the original pre-detection distribution of noise over the information bandwidth. The signal-to-noise ratio at the output of the decoder will be adequate (about 12 db) for automatic data processing.

8.5 Power and Power Program Systems Goddard Space Flight Center (F.C. Yagerhofer) (J. C. Schaffert)

The Electrical Power and Power Program Systems consist of a solar cell array, a storage battery, a solar array voltage regulator, a regulator converter, a program switch and recycle timer system, and a current sensing device.

The storage battery is a 13 cell, sealed, silver-cadmium system constructed in two separate containers, one of which contains the current sensing device. The latter is simply to monitor the charge and discharge of the battery system and supply this information to the telemetry on the performance parameter channel 15, frames 9 and 10. The current delivery capacity of the battery system is 4 amperehours. The average power requirement of the spacecraft is 16 watts. Typical spacecraft loads range from approximately 19.5 v at 700 ma to 12.8 v at 1400 ma.

The battery system is supplemented by a solar cell array mounted on paddles. The array consists of 5600 p-n junction silicon cells wired in a series-parallel arrangement to supply approximately 16 watts of electrical power at a potential of 19.5 volts.

The battery system is protected by a solar array voltage regulator circuit which clamps the charging voltage at a predetermined level by dumping excess power into a combination transistor and resistor load to be dissipated as heat.

The Program Switch and Recycle Timer system includes an undervoltage sensing circuit. When the voltage available from the solar array and battery combination falls below 12.8 volts as a result of either solar cell degradation or excessive battery drain, the regulator converter is switched off from the power source by the program switch and held off for a period of 8 hours, when the recycle timer will attempt to command the program switch to restore the load. If the solar array has restored the battery capacity in this period, the load will be switched on. If the battery has not been re-charged sufficiently, the recycle timer will command another 8 hour re-charging period. Both the program switch and recycle timer have been designed with redundant circuitry for high reliability.

The main regulator converter supplies the following voltages to the spacecraft:

6.5 VDC ±5 percent	-17.8 VDC ±1 percent
12.0 VDC ± 1 percent	26.2 VAC ±5 percent(@2KC ± 200 ~)

8.5.1 Telemetry Antenna Goddard Space Flight Center (J. K. Steckel)

The telemetry antenna for the S-3 payload is a modified crossed dipole turnstile. The elements are fed from an RF coaxial harness that incorporates a hybrid ring to divide the power and line lengths to phase the elements. The power patterns are nearly omnidirectional and are polarized circular on the spin axis and linear on a plane through the equator of the payload.

8.5.2 Telemetry Transmitter Goddard Space Flight Center (D. S. Hepler)

The telemetry transmitter is operated at a frequency of 136.020 Mc with a nominal output power of 2 watts. A pulsed frequency telemetry encoder signal is used to phase modulate the transmitter. Since this modulation is in the form of a pulsed square wave, a phase deviation of 57 degrees provides an equal amount of power in the carrier and each of the first two sidebands during the tone burst. The oscillator operates at one-half the carrier frequency. A buffer amplifier is used between the oscillator and phase-modulator to keep the incidental frequency modulation to a very low value, this provides a more suitable signal for use with phase-lock receiving systems in the ground stations. After the modulator the signal frequency is doubled and amplified to the 2-watt level. The transmitter operates from a single -17.8 volt source and draws approximately 400 milliamperes of current. A solid-state dc to dc converter is used to provide proper operating voltages for the final amplifier.

8.6 Trajectory (Goddard Space Flight Center)

8.6.1 Optimum Trajectory Requirements

Inclination Angle	33°
Apogee	50,000 ± 10,000 nautical miles
Perigee	$150 \pm {}^{100}_{0}$ nautical miles

In view of spacecraft weight, rocket performance tolerances, range safety and other factors, the absolute minimum requirements for trajectory are:

Apogee (93 percent probability)	40,000 nautical miles minimum
Perigee	150 nautical miles minimum

8.6.2 Time of Launch Restrictions

To insure minimum one-year lifetime with respect to effects of solar and lunar perturbations, as well as atmospheric drag, on the perigee altitude.

To limit extremes of temperature on sensitive portions of satellite by controlling angles of spin axis of satellite with respect to earth-sun line.

To consider avoidance of background radio noise from galactic hot spots while receiving telemetry and tracking data.

By combining calculated spin axis restrictions with solar-lunar perturbation restrictions, a launch window of approximately 2 hours was computed for a period of several weeks about the scheduled date of launch.

8.6.3 Satellite Time in Radiation Belts

Approximately 10 percent in V.A. count regions above 1000 c/s

Approximately 2.5 percent in V. A. count regions above 10,000 c/s

8.6.4 Satellite Time in Eclipse and in Sunlight

Approximately 98 percent of time in sunlight.

8.7 Tracking and Data-Acquisition Goddard Space Flight Center (Mr. J. J. Madden)

Owing to the highly eccentric orbit and to the anticipated quality and amount of data, it is expected that 1-2 weeks will be required to determine an accurate orbit. An attempt will be made to use the Trinidad radar to cover the point of injection. However, owing to the low horizon aspect of the target at this time, i.e., approximately 4 degrees elevation, it is not expected to acquire any precise injection velocity information other than an indication of successful orbit. Improved positional information will be developed from succeeding passes of the satellite.

8.7.1 Telemetry Operations

Because of the highly eccentric orbit, the spacecraft will be visible for approximately 23 hours at stations on the apogee side of the earth. Three receiving stations properly equipped and spaced in longitude will record the telemetry signal for 90 percent of the time. These stations are at Woomera, Santiago, and Johannesburg. These telemetry receiving antennas have 22 db gain, and will be circularly polarized. Other stations used for telemetry reception have approximately 19.2 db of gain, linear polarization. All stations are tuned to 136.020 Mc. Present plans are to record telemetry continuously for 1 month, and periodically thereafter as required by the Project Scientist. The expected life time of the spacecraft is one year minimum.

8.7.2 Tracking Operations

Existing Minitrack stations on the apogee side are being modified to improve signal-noise ratios on their fine-track antenna groups. Coarse and medium track antenna groups are not to be modified in time for this launch. The apogee tracking stations are at Woomera, Johannesburg, Santiago, Antofagasta, and Lima.

The rapid passes of the satellite through the perigee will be sampled as the satellite crosses the existing Minitrack stations such as Blossom Point, Maryland, Ft. Myers, Florida, Goldstone, California.

The orbital period is approximately 31 hours. On the first orbital pass the latitude is predicted to be 14°N at perigee, the longitude 49°W. At apogee, the latitude will be 12°S, the longitude 99°W.

At launch, early tracking data will be collected by the Ft. Myers, Florida Minitrack station and the GSFC Cape Canaveral T/M station. Azusa and other Atlantic Missile Range radar information will be available to GSFC for incorporation into a computer program. The vector information on range and velocity of the booster stages provided by the AMR radar track will provide inputs for computing the initial injection point velocities of the spacecraft to a nominal orbit, after which data from the Minitrack and other tracking stations will be added into the computer problem to correct the orbit calculations. As the Minitrack stations receive additional data, the accuracy of the calculated parameters of the orbit will be continuously improved.

The Ascension Island T/M receiver will provide an opportunity to acquire a first look at the spacecraft in orbit. AFMTC, AM Range will add a 136.020 Mc feed to the TLM-18 antenna there. The detailed Operations Plan for tracking and telemetry has been prepared by the Operations Control Branch (GSFC) and is titled S-3 Energetic Particles Satellite Operations Plan.

8.8 Data Reduction Goddard Space Flight Center (Mr. C. J. Creveling)

8.8.1 Equipment

The system that will be used for processing the data from S-3 is shown in Figure 5. The equipment was designed to handle all telemetry formats of this general category (PFM). The elements of the system are the Tape Converter-Comb Filter, PFM Digitizer-Computer Format Control Buffer, High Speed Line Printer, and CDC-160 Data Processor.

8.8.1.1 Tape Converter - Comb Filter

This portion of the system is designed to recover the telemetry signal in the presence of noise by utilizing the comb filter for signal-to-noise improvement and to recover the burst rate for use in synchronization. The comb filter has 120 filters equally spaced across the used frequency band with their response curves intersecting at the 3 db points. Integral logic permits determination of which filter is responding and

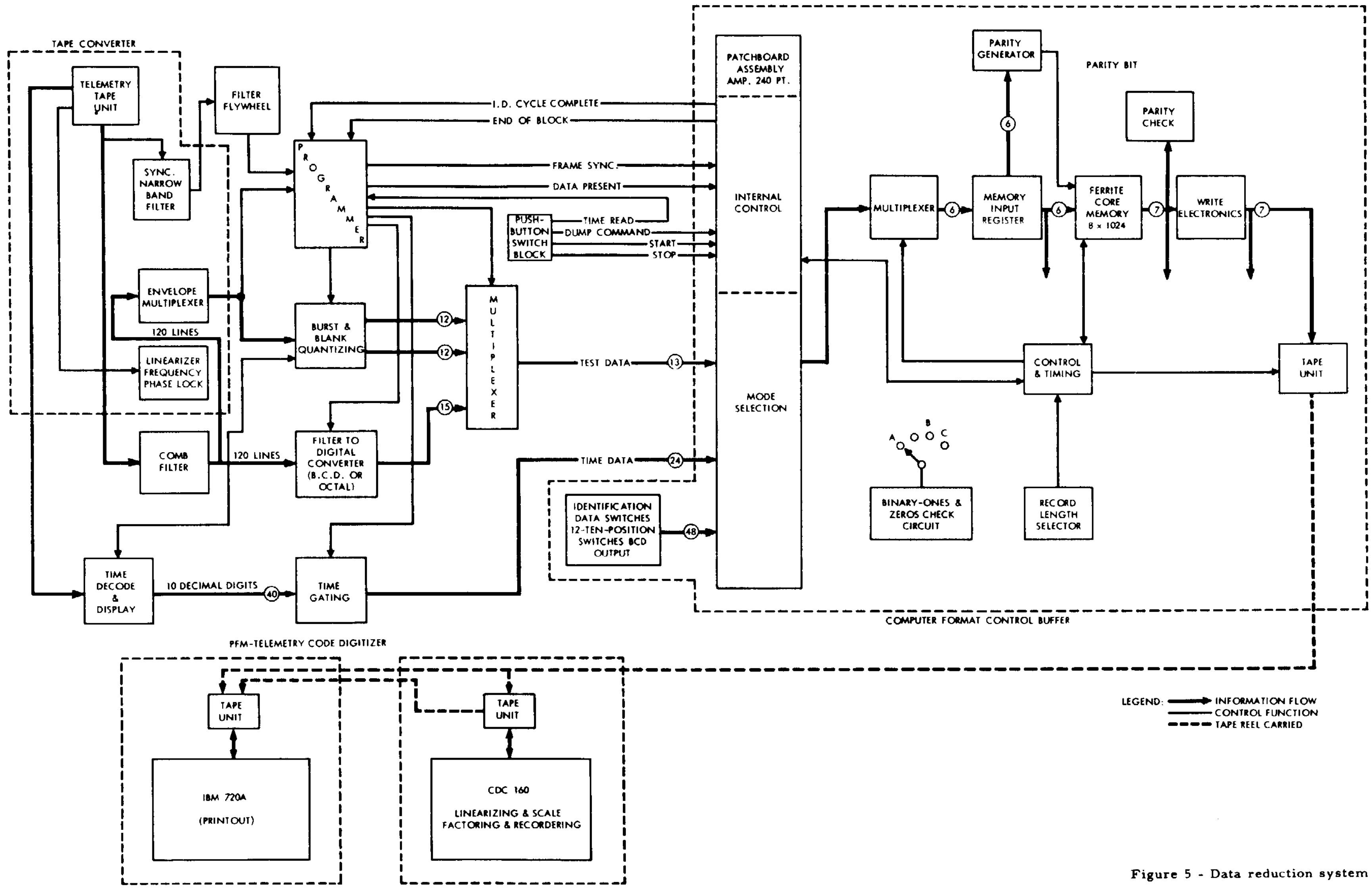


Figure 5 - Data reduction system

allows electronic removal of all other filters. In this way sufficient resolution is obtained to permit the filter to function directly as the frequency measurement device. This design assumes that a single frequency exists in each burst. Some departure from this condition can be tolerated.

8.8.1.2 PFM Digitizer - Computer Format Control Buffer

This equipment will utilize the outputs of the above equipment in such a manner as to establish synchronization, encode the 120 lines from the comb filter, and multiplex the frequency data with the time code. Time is stored from the time decoder when a frame synchronization occurs. The output of the Format Control Buffer is a digital magnetic tape in IBM binary coded decimal format suitable for further processing by computer, or off-line printer. The frequency data will appear as a number between 0 and 119 in the case of analog channels and as a number between 0 and 7 in the case of digitally encoded channels. In the latter case this permits retention of the original bit configuration so that a full digital word can be accumulated visually on print-out or by the computer.

8.8.1.3 High Speed Line Printer

The digital magnetic tapes prepared on the Format Control Buffer can be printed on this device. A full dataprint-out in terms of the telemetry and system units with one frame per line plus time of the synchronization pulse will be the result. Printing of selected sections of the full format can be provided after a simple operation in the off-line data processor.

8.8.1.4 CDC-160 Data Processor

The CDC-160 Data Processor is a small transistorized stored program digital computer with magnetic tape and control equipment. The output magnetic tapes from the above equipment can be used as an input. Programs are available to decommutate, edit, accumulate, and record the data and provide digital magnetic tapes for further use as required.

8.8.2 Procedure

The station telemetry tapes containing a 100 kc standard frequency, Minitrack time code, telemetry signal, etc. will be received by the Data Systems Design Branch for editing and storage. Each tape will be reviewed for quality and quantity of useable data and checked against expected station performance. A summary of tapes by station will be maintained as a control input for station operation and as a guide for processing. It is anticipated that three categories of tapes will be stored. They are (1) unuseable tapes resulting from inadequate signal-to-noise ratio, interference, or operator error, (2) questionable tapes requiring extra handling to recover the data, and (3) good tapes of sufficient quality to warrant immediate machine processing. The first class of tapes will be retained for archival purposes and possible exploitation as the state of the art is advanced. The remaining two categories will be processed through the system described. Approximately three analog tapes will be placed on one digital tape and the originals stored for future reference. The specific handling will depend upon the quality of the recorded signal. The criterion will be the reliability of synchronization determination as the digital tapes are made.

8.8.2.1 Good Tape

These tapes are characterized by a good signal-to-noise ratio and consequently a high reliability of synchronization determination as mentioned above. This condition will permit preparing the digital format around the known synchronization location so that a full data printout can be made immediately without calibration or linearization, data sequence correction or grouping, or error correction and removal. The frequency information will appear as previously described. Calibration information will be provided. This method will permit a quick look at the data. Further processing will be accomplished in the CDC 160 Data Processor. It will accept the magnetic tape prepared with time. The data will be decommutated by experiment and placed on magnetic tape in a form suitable for printing on an IBM 720A line printer. The output will be one frame per line, columnized by channels with one column for time or an assembled digital word with a suitable time reference as required. Identification data will be inserted as the first record of each digital magnetic tape. It will include day of the year, station number, experiment number, and analog tape number. This information will be printed at the top of each page.

8.8.2.2 Questionable Tapes

The extra handling is a direct result of poor signal-to-noise ratio. In this case special processing in a large digital computer will be necessary. This method is very time consuming and hence costly and should be avoided when possible. Adequate signal-to-noise ratio is the best answer.

A number of modifications are possible as operating procedures are improved and experience is gained. However, it is anticipated that no serious departures from the program outlined will be required. Alternate and additional output forms such as punched cards can be provided where required. However, it should be noted that many of these require conversions from digital magnetic tape and as such are slow down processes. Each such slow down adds to the overall time of processing for the experiment in question. Every effort should be made to use the digital magnetic tape where possible.

8.9 Spacecraft Test Stand Equipment Goddard Space Flight Center (C. J. Creveling)

The equipment of the S-3 test stand is divided into five major categories:

Transducer Simulators - and simulated sources, such as radioactive and light sources to energize transducers of some spacecraft experiments; current inputs and count inputs are used as simulators for other experiments.

Spacecraft Control and Monitoring Equipment - such as blockhouse control unit, external power supplies, and high accuracy voltmeters.

Signal Receiving and Storage - included here are equipment for receiving, displaying, and demodulating the phase modulated carrier, and for RF frequency and power measurements, as well as WWV time signals, and tape recorders.

PFM decoding to extract channel, frame, and master frame synchronization signals, and generate gates to select any desired channel for examination and printout.

Experiment Logic Decoding - collects PFM-to-digital decoder outputs, restores original format and converts to equivalent decimal form. Programmers are used to identify the experiments being transmitted, select proper conversion format, and provide inputs to printout equipment.

8.10 Third Stage Spacecraft Operation Sequence

Time (secs)	
T - 2	Spin up third stage to 150± rpm. 1200 sec (GSFC) spin actuated timer starts.
T - 0	Separate third stage from second stage and ignite third stage (X248). Start 1500 sec (DAC) timer.
T + 43	Third stage burnout.
T + 1198	1200 sec (GSFC) timer actuates and deploys yo-yo despin device to slow third stage and spacecraft to about 31 rpm.
T + 1500	1500 sec timer (DAC) actuates paddle release guillotines and starts 2 sec time delay relay (DAC). Paddle erection despins third stage and spacecraft to 18 ± 2 rpm.
T + 1502	Spacecraft separates from third stage with velocity of 6 f.p.s., on signal from 2 sec delay relay which fires explosive bolt to release Marmon Clamp.

(GSFC) Refers to timers or relays within S-3 spacecraft.

(DAC) Refers to timers in third stage.

9. DIRECTIVES

None available at this time.

APPENDIX B

PROJECT DEVELOPMENT

APPENDIX B: Project Development

Project authorized	May, 1959
Engineering effort on project started	February, 1960
Design completed	August, 1960
Prototype Unit assembled and integration completed	March, 1961
Prototype Unit testing started	March 22, 1961
Flight Spacecraft assembled and integration completed	April, 1961
Flight Spacecraft testing started	May 20, 1961
Prototype Unit testing completed	May 31, 1961
Flight Spacecraft testing completed	June 14, 1961
Flight Spare Spacecraft assembled and integration completed	June, 1961
Flight Spare Spacecraft testing started	June 19, 1961
Prototype Unit retests	June 25, 1961
Flight Spare Spacecraft testing completed	July 7, 1961
Flight Spacecraft at AMR	July 10, 1961
Launch	August 15, 1961

APPENDIX C

TEST PLAN - STRUCTURAL PROTOTYPE

TEST PLAN
FOR
STRUCTURAL PROTOTYPE
ENERGETIC PARTICLES SATELLITE
(S-3)
DELTA PAYLOAD

BY:

F. A. CARR

OCTOBER 24, 1960

TEST AND EVALUATION DIVISION
OFFICE OF TECHNICAL SERVICES
GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

C-1

APPROVED BY:

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J. M. Madey	Structural Design
Code 326	
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F. Carr)	Coordinators
H. Maurer	Alternate
Code 321	
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N. C. Schaller	Acceleration, Balance

1.0 INTRODUCTION

The Energetic Particles Satellite (S-3) will be launched by a Delta vehicle which employs a Douglas-Thor liquid-fuel first stage, an Aerojet-General AJ10-118 liquid-fuel second stage, and an ABL X-248-A5 solid-propellant third stage.

The Structural Prototype is a model of the intended S-3 payload structure. It contains no electronic equipment.* The intended flight subassemblies are simulated with respect to weight and size by dummy weights placed in approximate positions in the model. Thus, a first approximation to a dynamically similar payload is achieved.

Since no electronic equipment is installed in the Structural Prototype, no monitoring during exposures is required.

This environmental test program is intended to serve as a means of gaining information which may be useful for subsequent prototype and flight payload design and/or test.

For the purposes of this document, the "three major axes" of the Structural Prototype shall be defined as follows:

Z-Z or Thrust Axis: A line passing through the CG of the Prototype parallel to the thrust axis of the vehicle.

Transverse Axes: Two mutually perpendicular lines, passing through the CG of the Structural Prototype, and perpendicular to the thrust axis. One transverse axis shall be selected so that vibration or acceleration along this axis would be most likely to produce a failure. The second axis shall be perpendicular to this axis.

2.0 SCOPE

This document supersedes the test specification dated October 3, 1960, titled: "Test Program for Structural Model, Energetic Particles Satellite."

* A Telemetry Encoder has been installed. No monitoring required.

This specification applies to the S-3 Structural Prototype. A later document will specify the environmental test programs for S-3, subassemblies, Prototype Payload, Flight Payload, and Flight Spares.

The Structural Prototype will be exposed to the environments of vibration, acceleration, and spin, after having been statically balanced and the center of gravity determined.

3.0 VIBRATION

The Structural Prototype will be mounted to a fixture simulating the actual attachment of the payload to the X-248 motor. The paddle arms will be in the folded position and secured in a manner which is similar to that expected to be used during actual powered flight.

The duration of each vibration test shall be recorded.*

Input acceleration shall be measured as near to the interface between the fixture and the structural prototype as possible.

3.1 Procedure

3.1.1 Transmissibility Test (Including Flight and Sinusoidal Sweep Tests)

The transmissibility test shall consist of three parts.

3.1.1.1 Part I:

While exposing the Structural Prototype to an acceleration input of 1g-vector, sinusoidal sweep, a probing survey shall be conducted to determine regions of high amplification. The probing shall be done in accordance with a predetermined plan, in order that the length of time required for an adequate transmissibility study is minimized.

* The natural frequency and amplification of the fixture shall be recorded.

The sweep shall be interrupted to permit probing only at resonant frequencies and 600 cps.

With the advice of the design group and the information obtained regarding regions of high amplification, acceleration pickups shall be mounted at critical locations.

In addition to recording the duration of vibration, the time spent at each resonant frequency shall be recorded.

3.1.1.2 Part II:

A second sinusoidal sweep shall be conducted in accordance with the "Flight Payload" levels listed below. By means of continuous monitoring of the acceleration pickups, a second point on the transmissibility curve can be obtained without interrupting the sweep.

Frequency Range cps	Amplitude			
	Thrust Axis		Transverse Axes	
	g-rms	g-vector	g-rms	g-vector
5-50	1	1.5	0.4	.6
50-500	5	7	1	1.5
500-2000	10	14	2	2.8
2000-5000	40	56	8	11

Notes:

(1) Amplitude limited to 0.5" peak to peak.
(2) The sweep rate shall be four octaves/min.

3.1.1.3 Part III:

A third sinusoidal sweep shall be conducted in accordance with the "Prototype" payload levels listed below. Continuous monitoring will indicate a third point on the transmissibility curve.

Frequency Range cps	Amplitude			
	Thrust Axis		Transverse Axes	
	g-rms	g-vector	g-rms	g-vector
5-50	1.5	2.12	0.6	0.85
50-500	7.5	10.3	1.5	2.12
500-2000	15	21.2	3	4.25
2000-5000	60	84.8	12	16.9

Notes:

(1) Amplitude limited to 0.5" peak to peak.
(2) The sweep rate shall be approximately two octaves/min.

Each of the procedures listed in Parts I through III shall be conducted for each of the Structural Prototype's three orthogonal axes.

3.1.2 Random (Spectral density at twice Flight Payload level)

Levels of exposure shall be in accordance with the following table for each of three major axes:

Frequency Range	Spectral Density	Amplitude $g = \sqrt{\Delta f \ g^2/cps}$	Duration
20-2000 cps	0.1 g^2/cps	15.4 g-rms	* 4 Min.

* In each direction (total time - 12 minutes).

3.1.3 X-248 Resonant Burning (600 cps Sinusoidal Vibration)

Levels of exposure shall be in accordance with the following table:

Axis	Frequency	Force (Pounds)	Duration (Min.)
Thrust Axis	550-650	± 600 *	0.5
Transverse Axes	550-650	± 100 **	0.5

* Corresponds to 86 g-rms for a payload having an apparent weight of 5 pounds as determined at 600 cps.

** Corresponds to 15 g-rms for a payload having an apparent weight of 5 pounds as determined at 600 cps.

4.0 ACCELERATION

4.1 Procedure

4.1.1 Thrust Axis

The Structural Prototype will be exposed to acceleration along the thrust axis in the forward direction. The magnitude shall be 28g, as measured at the CG, for a duration of one minute.

5.0 SPIN

5.1 The Structural Prototype will be subjected to a spin rate of 180 rpm about its thrust axis for 1-1/2 minutes. The paddles will be attached to the Structural Prototype in the folded position during spin.

APPENDIX D

ENVIRONMENTAL TEST PLAN

ENERGETIC PARTICLES SATELLITE

(S-3)

APPENDIX D

ENVIRONMENTAL TEST PLAN

ENERGETIC PARTICLES SATELLITE

(S-3)

PREPARED BY:

PAUL TARVER
HENRY MAURER
FRANK CARR

FEBRUARY 14, 1961

TEST AND EVALUATION DIVISION
OFFICE OF TECHNICAL SERVICES
GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ENVIRONMENTAL TEST PLAN
ENERGETIC PARTICLES SATELLITE
(S-3)

Approved By	Code	Date
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/S/ H. E. LaGow H. E. LaGow Chairman, Environmental Committee	600	3/1/61
/S/ J. W. Townsend, Jr. J. W. Townsend, Jr. Assistant Director, SSSA	600	3/2/61

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ENVIRONMENTAL TEST PLAN
ENERGETIC PARTICLES SATELLITE
(S-3)

1.0 SCOPE

The purpose of this document is to specify an environmental test program for subassemblies and spacecraft systems of the Energetic Particles Satellite (S-3).

2.0 INTRODUCTION

2.1 Satellite Description

The S-3 will be placed in a highly eccentric orbit. Its distance from the earth will range from 150 miles at perigee to a minimum apogee of 40,000 miles. This orbit will be particularly advantageous for the study of the physics of fields and energetic particles in space.

The satellite will observe the solar wind, the interplanetary magnetic field, and the distant regions of the earth's magnetic field. In addition, it will observe the particle population of interplanetary space and the trapped radiation regions, measuring particle fluxes, types, and energies as functions of position, direction, and time.

S-3 will be launched by a Delta vehicle which employs a Douglas Thor liquid-fuel first stage, an Aerojet-General AJ10-118 liquid-fuel second stage, and an ABL X-248-A-5 solid-propellant third stage.

2.2 General Testing Philosophy

One Prototype Payload and one Flight Payload will be constructed. In addition, a third system ("Test Set") will be fabricated to house each of the two sets of flight spares (separately) for environmental qualification.

Four units of each subassembly will be manufactured--one for installation in the prototype system and one for the flight system. The remaining two will be flight spares, and each set will be separately installed in the "Test Set" system.

Environmental exposures and appropriate operational tests will be conducted on S-3 subassemblies and spacecraft systems. The levels of exposures and tests specified herein were determined by consideration of the anticipated environments induced by ground handling and storage, the launch vehicle, and the expected space environment the spacecraft will experience. Two levels of exposures are specified--one for <u>Prototype Qualification</u> and one for <u>Flight Acceptance</u>.

The Prototype Qualification test program provides assurance that the design is capable of withstanding expected flight environment, as well as conditions encountered during shipment, storage, and handling. These environmental exposures are more severe than field conditions, in order to provide greater assurance for detecting and locating faults in the system. The environmental levels specified, however, are not so severe as to exceed reasonable design safety margins or to excite unrealistic modes of failure.

The Flight Acceptance program is specified for the flight spacecraft system, space instrumentation, and flight spares. These environmental exposures will be at levels equal to expected flight and field conditions. They will (1) provide assurance that the units to be actually flown will withstand flight environment and (2) by testing a second system, provide additional assurance of reliability at flight levels.

The environmental testing philosophy and sequence indicated above can be summarized as follows:

1. Qualification of Prototype Subassemblies
2. Qualification of Prototype System
3. Acceptance of Flight Subassemblies
4. Acceptance of Flight System
5. Acceptance of Set 1 - Flight Spares*
6. Acceptance of Set 2 - Flight Spares*

    * Through environmental test of "Test Set System".

This method of flight spare acceptance has no provisions for proving interchangeability of the flight spares with the flight units. To prove interchangeability completely, an impractical number of tests would be required due to the number of combinations possible.

In order to arrive at some level of confidence of interchangeability, the following procedure will be followed, schedules permitting:

1. After the flight payload system and the test set system, containing flight spares, have been environmentally tested and accepted, the flight model encoder shall be installed in the test set system and therein exposed to thermal vacuum in accordance with paragraph 5.7.3. The success of this test will verify interchangeability of all the spare subassemblies (set 1) with the flight encoder.

2. In a similar manner, the flight model encoder will also be installed in the test set system, containing the spare (set 2) subassemblies, and therein exposed

to thermal vacuum in accordance with paragraph 5.7.3. The success of this test will verify interchangeability of all the spare subassemblies (set 2) with the flight encoder.

This procedure will verify the interchangeability of the flight model encoder and all the spare subassemblies in the event that substitution of subassemblies is required.

2.3 Monitoring

Prototype and Flight Payloads, and/or subassemblies, shall be operated and monitored before and after or during environmental exposures, as applicable. The criterion to be applied is that those operations which will occur during the environment being simulated shall be operated during, as well as before and after, exposure. All other operations shall be operated before and after exposure. Where redundant circuits or components are used, each circuit or component shall be monitored independently.

It will be the responsibility of the project manager to provide a detailed Test Plan for the operation and monitoring of the payloads and subassemblies. This shall include a procedure for checking each function, operating parameters, a description of any special equipment required, and specifications for satisfactory payload and subassembly operation, including calibration levels and permissible deviations during and after environmental exposure.

It will be the joint responsibility of the project manager and the Test and Evaluation Division to provide monitoring equipment and personnel to operate and monitor payloads and subassemblies.

3.0 ENVIRONMENTAL EXPOSURES FOR SUBASSEMBLIES AND EXPERIMENTS

The following applies specifically to instrumentation subassemblies. The levels of exposure may be applied to experiment packages where applicable.

3.1 First Models (those intended for installation in the Prototype Payload)

The first-model subassemblies will be exposed to vibration, acceleration, and temperature at levels which are identical to those specified for the Prototype Payload system. Environmental levels and procedures are set forth in the following paragraphs:

Exposure	Paragraph
Vibration	5.4.1.1, 5.4.1.2
Acceleration	5.2
Temperature	5.6.1
Thermal-Vacuum	5.7.3

After completion of these exposures, the first models will be installed in the Prototype Payload. The payload will then be exposed to the entire series of environmental tests, as specified in paragraph 4.1.

3.2 Second Models (intended flight articles)

These second models will be exposed to vibration and thermal-vacuum at flight payload levels. Environmental levels and procedures are set forth in the following paragraphs:

Exposure	Paragraph
Vibration	5.4.2.1, 5.4.2.2
Thermal-Vacuum	5.7.3

3.3 Third and Fourth Models (flight spares)

The third set of subassemblies will be installed in the "Test Set" system and exposed to a Flight Acceptance environmental program similar to that specified in paragraph 4.2 for the Flight Payload.

NOTE: Table I of the appendix summarizes the procedure for the environmental testing of all subassemblies as described above.

4.0 ENVIRONMENTAL EXPOSURES FOR SPACECRAFT SYSTEMS

4.1 Prototype Payload

After statically and dynamically balancing, in accordance with restraints set forth in a separate document, the Prototype Payload will undergo the following environmental tests in the order listed:

1. Spin .... (Par. 5.1)
2. Acceleration .... (Par. 5.2)
3. Shock .... (Par. 5.3.1)
4. Vibration .... (Par. 5.4.1)
5. Humidity .... (Par. 5.5)
6. Temperature .... (Par. 5.6.1)
7. Thermal-Vacuum .... (Par. 5.7.1)

Applicable paragraphs indicating procedures and levels are shown in parenthesis.

4.2 Flight Payload

The Flight Payload will be exposed to vibration, shock, and thermal-vacuum. Applicable paragraphs giving procedures and levels are as follows:

Exposure	Paragraph
Vibration	5.4.2
Shock	5.3.2
Thermal-Vacuum	5.7.2

A static and dynamic balancing operation will be performed before and after environmental testing. The balance requirements and restraints shall be set forth in a separate document.

NOTE: Table II of the appendix summarizes the procedure for the environmental testing of payloads as described above.

5.0 ENVIRONMENTAL EXPOSURES AND TEST LEVELS

Before and after the environmental test program, the Prototype Payload and the Flight Payload shall be statically and dynamically balanced in accordance with restraints set forth in a separate document.

Special environmental tests which may be required and which are not included in this document will be contained in a supplement to this specification.

5.1 Spin

Prototype Qualification

The payload with the arms folded and secured shall be subjected to a spin rate of 180 rpm for a duration of one minute.

The payload shall be operative during the spin test.

5.2 Acceleration

Prototype Qualification

5.2.1 Subassemblies

The axial acceleration shall be 28g for one minute.

5.2.2 Prototype Payload

Axial and lateral acceleration shall be applied to the payload simultaneously. The paddles shall be in the folded position.

The acceleration, as measured at the top of the octagonal instrument package of the payload, shall be such that the axial component of the acceleration will be 28g, and the transverse component of the acceleration will be 3g.

The angular velocity of the rotating arm of the centrifuge shall be increased gradually until the desired acceleration is attained. This acceleration shall be held for approximately one minute and the angular velocity gradually decreased to zero.

The payload shall then be rotated 90° about the payload's thrust axis and the above procedure repeated.

A total of four tests shall be performed, each time rotating the payload by 90°.

5.3 Shock

The payload's paddle arms shall be in the folded position during the test.

The shock pulse shall be applied to the payload through the payload - X-248 mounting interface, in the thrust direction.

The values of the peak amplitudes contain both stage ignition and payload transportation considerations. The Delta ignition shock, however, is expected to be lower-than-expected transportation shock.

5.3.1 Prototype Qualification

Shock Pulse	Pulse Duration	Peak Acceleration, g
1/2 Sine Wave	11 Milliseconds	22.5

5.3.2 Flight Acceptance

Shock Pulse	Pulse Duration	Peak Acceleration, g
1/2 Sine Wave	11 Milliseconds	15

5.4 Vibration

The procedures for conducting the vibration exposures shall be in accordance with the Delta Payloads Testing Procedures of Code 321.2 (page D-19 of the appendix).

5.4.1 Prototype Qualification

5.4.1.1 Sinusoidal Vibration

Levels of exposure shall be in accordance with the following table:

Frequency Range (cps)	Vector Acceleration (g)	
	Thrust Axis	Transverse Axes
5-50	2.3	0.9 (a)
50-500	10.7	2.1
500-2000	21	4.2
2000-3000	54	17
3000-5000	21	17 (b)

NOTES:

(a) Within maximum amplitude limit of vibration generator.

(b) Within maximum frequency limit of vibration generator.

(c) The sweep rate shall be two octaves per minute.

(d) The duration of the exposure shall be approximately five minutes in each direction (total time - 15 minutes).

5.4.1.2 Random

Levels of exposure shall be in accordance with the following table for each of the three major axes:

Frequency Range (cps)	Spectral Density (g^2/cps)	Amplitude $g=\sqrt{\Delta f \; g^2/cps}$ (g-rms)	(a) Duration Min.
20-2000	.07	11.5	4.0

NOTES:

(a) Four minutes each axis - total time: 12 minutes.

(b) White Gaussian noise with g-peaks clipped at three times the rms acceleration.

5.4.1.3 X-248 Combustion Resonance

Levels of exposure shall be in accordance with the following table:

Axis	Frequency	Force	Duration
Thrust	550-650 cps	(a) ±600 pounds	30 Seconds
Transverse Axes	550-650 cps	(a) ±100 pounds	30 Seconds

NOTES:

(a) If it is not possible to program force with available equipment, the vector acceleration shall be determined by dividing ±600 lbs. force (thrust axis) or ±100 lbs. (transverse axes) by the apparent weight of the payload in pounds, as measured over the 550-650 cps range.

(b) Each test is conducted by sweeping once at a rate so that 30 seconds are required to traverse the band from 550 cps to 650 cps.

5.4.2 Flight Acceptance

5.4.2.1 Sinusoidal Vibration

Levels of vibration shall be in accordance with the following table:

Frequency Range (cps)	Vector Acceleration (g)	
	Thrust Axis	Transverse Axes
5-50	1.5	0.6 (a)
50-500	7.1	1.4
500-2000	14	2.8
2000-3000	36	11.3
3000-5000	14	11.3 (b)

NOTES:

(a) Within maximum amplitude limit of vibration generator.

(b) Within maximum frequency limit of vibration generator.

(c) The sweep rate shall be four octaves per minute.

(d) The duration of the exposure shall be 2-1/2 minutes each direction (total time - 7-1/2 minutes).

5.4.2.2 Random

Levels of exposure shall be in accordance with the following table for each of the three major axes:

Frequency Range (cps)	Spectral Density (g^2/cps)	Amplitude $g=\sqrt{g^2/cps \triangle f}$ (g-rms)	(a) Duration Min.
20-2000	.03	7.7	2.0

NOTES:

(a) Two minutes each axis - total time: 6 minutes.

(b) White Gaussian noise with g-peaks clipped at three times the rms acceleration.

5.4.2.3 X-248 Combustion Resonance

Levels of vibration shall be in accordance with the following table:

Axis	Frequency	Force	Duration
Thrust	550-650 cps	(a) ±400 pounds	15 Seconds
Transverse Axes	550-650 cps	(a) ± 50 pounds	15 Seconds

NOTES:

(a) If it is not possible to program force with available equipment, the vector acceleration shall be determined by dividing ±400 lbs. (thrust axis) or ±50 lbs. (transverse axes) by the apparent weight of the payload, as measured over the 550-650 cps range.

(b) Each test is conducted by sweeping once at a rate so that 15 seconds are required to traverse the band from 550 cps to 650 cps.

5.5 Humidity

Prototype Qualification

While non-operative the payload shall be subjected to a test chamber temperature of 40°C with a relative humidity of 95% for 50 hours. The chamber temperature shall then be lowered to 25°C with the relative humidity maintained at 95%. The equipment shall then be operated and its performance checked.

5.6 Temperature

The in-flight temperature of the satellite, with power on, is expected to range from 0°C to +40°C. The operative temperature test levels (par. 5.6.1.2) include a 10°C safety factor.

5.6.1 Prototype Qualification

5.6.1.1 Non-Operative Test

While non-operative the subassembly or payload shall be subjected to a test chamber temperature of -30°C for six hours, followed by a temperature of +60°C for six hours.

5.6.1.2 Operative Test

The subassembly or payload shall be operative and the temperature of the chamber lowered so that the subassembly or a selected point in the payload attains a temperature of -10°C. The equipment shall be checked after the temperature has stabilized at -10° ±2°C.

The temperature of the chamber shall then be raised until the subassembly or a selected point in the payload attains a stabilized temperature of +50°C ±2°C. The performance of the equipment shall then be checked.

5.7 Thermal-Vacuum

5.7.1 Prototype Qualification

5.7.1.1 High Temperature Test

1. Vacuum - The chamber shall be evacuated to a pressure of 1×10^{-4} mm Hg or less with the payload or subassembly remaining at simulated launch temperature.

2. Temperature - The walls of the chamber and other radiant sources shall be maintained at a temperature such that a selected point in the payload (while operative) attains a temperature of +50°C. The temperature distribution shall be reasonably representative of actual flight conditions.

3. Duration - The test shall last a minimum of seven days of continuous duty-cycle operation.

5.7.1.2 Low Temperature Test

1. Vacuum - The chamber shall be maintained at a pressure of $1X10^{-4}$ mm Hg or less.

2. Temperature - The walls of the chamber shall be maintained at a temperature such that a selected point in the payload (while operative) attains a temperature of -10°C. The temperature distribution shall be reasonable of actual flight conditions.

3. Duration - If this low temperature test can be accomplished as a direct continuation of the previous high temperature test, without the chamber being opened, then the duration shall be three days. The payload will be maintained for at least two of the three days at the temperature described in paragraph 2., above.

If the chamber is opened, the test shall last five days from the time stable conditions are established.

5.7.2 Flight Acceptance

5.7.2.1 High Temperature Test

1. Vacuum - The chamber shall be evacuated to a pressure of $1X10^{-4}$ mm Hg or less with the payload remaining at simulated launch temperature.

2. Temperature - The walls of the chamber shall be maintained at a temperature such that a selected point in the payload (while operative) attains a minimum temperature of +40°C. The temperature distributions shall be representative of actual flight conditions.

3. Duration - The test shall last three days.

5.7.2.2 Low Temperature Test

1. Vacuum - The chamber shall be maintained at $1X10^{-4}$ mm Hg or less.

2. Temperature - The walls of the chamber shall be maintained at a temperature such that a selected point in the payload (while operative) attains a minimum temperature of 0°C. The temperature distribution shall be reasonably representative of actual flight conditions.

3. Duration - If this test is a direct continuation of the previous high temperature test, without the chamber being opened, then the duration shall be two days. If the chamber is opened, the test shall last three days from the time stable conditions are established.

5.7.3 Short Duration Tests

The duration of the thermal-vacuum tests for subassemblies shall be 24 hours for the high temperature test and 24 hours for the low temperature test. All other provisions of paragraphs 5.7.1 and 5.7.2 are unchanged.

The duration of the thermal-vacuum tests for the Test Set System with the flight model encoder installed shall be 24 hours for the high temperature test and 24 hours for the low temperature test. All other provisions of paragraphs 5.7.1 and 5.7.2 are unchanged.

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TABLE 1

S-3 SUBASSEMBLY ENVIRONMENTAL TESTS AND LEVELS

Subassembly	Spin Balance		Acceleration		Shock		Vibration		Humidity		Temperature		Thermal Vacuum	
Levels →	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL
1. First Model (intended for PT/PL)														
a. Transmitter, Convertors, Command Receiver, etc.			X				X				X			
b. Batteries			X				X				X		X	
c. Encoder			X				X				X			
2. Second Model For Flight Payload														
a. Trans., Rec., etc.								X						X
b. Batteries								X						X
c. Encoder								X						X

Key: PT/PL - Prototype Payload Levels
F/PL - Flight Payload Levels

TABLE II

SYSTEMS ENVIRONMENTAL TESTS AND LEVELS

	Bal-ance	Spin		Vibration		Accel-eration		Thermal Vacuum		Temperature		Humidity		Shock	
		PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL	PT/PL	F/PL
1. Prototype Payload	X	X		X		X		X		X		X		X	
2. Flight Payload	X				X				X						X
3. Test Set System containing 1st Set of Flight Spare Subassemblies	X				X				X						X
4. Test Set System containing 1st Set of Flight Spare Subassemblies, except with Flight Model Encoder replacing 1st Spare Encoder									X						
5. Test Set System containing 2nd Set of Flight Spare Subassemblies	X				X				X						X
6. Test Set System containing 2nd Set of Flight Spare Subassemblies, except with Flight Model Encoder replacing 2nd Spare Encoder									X						

Key: PT/PL - Prototype Payload Levels
F/PL - Flight Payload Levels

DELTA PAYLOAD VIBRATION TESTING PROCEDURES

1. Installation for Test

The payload shall be attached to a vibration generator via a rigid jig fabricated with payload adapter duplicated from the rocket motor design. The payload should be operated electrically, as would be the case in actual launch or flight, and monitored for malfunctions in telemetering or other systems which yield data during boost. Antennae or other dual-position devices shall be in proper position relative to the vehicle program sequence with the boost stage where a particular vibration is introduced. For this purpose, sinusoidal vibration shall be considered to occur during X-248 motor operation; random vibration during launch and maximum dynamic pressure flight. For purposes of controlling vibration applied to the payload, a calibrated accelerometer shall be attached rigidly on the jig near the payload-jig interface and trued with the axis of applied vibration. In addition, two other accelerometers, with sensitive axes mutually perpendicular with the first, shall be located near the interface (on the jig) to monitor uncontrolled lateral cross-talk.

2. Sinusoidal Vibration Records

The three accelerometer signals shall be recorded continuously during all sinusoidal tests. Due care shall be exercised to calibrate the overall system for frequency response and amplitude linearity characteristics to values 1.25 times the maximum expected to be recorded during tests. Permanent records then shall be made, properly labeled, and retained to demonstrate conformance with this specification.

3. Random Vibration Records

During the random vibration tests, signals from the control accelerometer shall be passed through a bandpass-filter-type analyser which has been adjusted to scan the test spectrum in the applicable test-duration time. The filter bandwidth shall be as narrow as allowed by the testing time and the length of the spectrum to be traversed. Permanent records shall be made during the specified random vibration tests, properly labeled, and retained to demonstrate conformance with this specification.

APPENDIX D: ATTACHMENT NO. 1

April 5, 1961

MEMORANDUM FOR HEAD, System Evaluation Branch
Test and Evaluation Division

Subject : Temperature and Humidity Tests for S-3 Prototype System Payload

Reference: (a) Environmental Test Plan Energetic Particles Satellite (S-3), dated February 14, 1961, Test and Evaluation Division

(b) Memorandum dated March 16, 1961, from Thermal-Vacuum Test Section to Thermodynamics Branch

Subject tests are attached as Enclosure 1. They supersede the tests described in references (a) and (b).

/s/ A. R. Timmins
A. R. Timmins, Head
Thermal-Vacuum Test Section
Thermodynamics Branch

Enclosure

ART:kmh

cc: Office of Technical Services (1)
Thermodynamics Branch (10)
Thermal-Vacuum Test Section (4)
Thermal-Vacuum Facilities Section (4)
Electronics Test Branch (2)
System Evaluation Branch (40)
Thermal Systems Branch (2)

TEMPERATURE AND HUMIDITY TESTS FOR S-3 PROTOTYPE SYSTEM PAYLOAD

Temperature (Non-Operative):

While non-operative, the payload shall be subjected to a test chamber temperature of -10°C (±2°C) for six hours followed by a temperature of 50°C (±2°C) for six hours. The payload temperature is then reduced to 30°C (±2°C) and the system operated.

Temperature (Operative):

While non-operative the payload temperature is lowered and stabilized at -10°C. Stabilization is indicated when the temperature between the second and third stacks of the main encoder is maintained at -10° ±2°C for at least one hour. The payload is then operated to demonstrate the functioning of each experiment. Operation is continued until the temperature of each experiment has stabilized (temperature does not change more than 0.5°C in an hour).

With the payload in the non-operative state, the temperature is raised until the payload is stabilized at 35°C. Stabilization is indicated when the temperature between the second and third stacks of the main encoder is maintained at 35° ±2° for at least an hour. The payload is then operated to demonstrate the functioning of each experiment. Operation is continued until the temperature of each experiment has stabilized. (Temperature does not change more than 0.5°C in an hour).

Humidity Test:

While non-operative, the payload shall be subjected to a test chamber temperature of 30°C and a relative humidity of 95%. Operational checks (sufficient to demonstrate operability) shall be made after two and four hours with chamber conditions maintained at 30°C and 95% RH. The exposure is continued for a minimum of 16 hours. At the end of the exposure period, an operational check is made followed by a complete system check-out.

APPENDIX D: ATTACHMENT NO. 2

March 30, 1961

MEMORANDUM FOR Head, System Evaluation Branch
Test and Evaluation Division

Subject : Thermal-Vacuum Test Procedure for the S-3 Prototype System

Reference: (a) Memorandum dated March 9, 1961 to Head, Thermodynamics Branch, subject: "Proposed Thermal-Vacuum Test Procedure for S-3 Prototype System"
(b) Environmental Test Plan, Energetic Particles Satellite (S-3) dated February 14, 1961

The subject test procedure is attached. It supersedes the test plans given in references (a) and (b) for the thermal-vacuum test.

The test procedure has been reviewed and coordinated with the Program Manager, the Electronics Test Branch, and the System Evaluation Branch.

Test No. 4, the cold soak thermal-vacuum test, will be conducted at both 0°C and -10°C. Zero degrees Centrigrade is considered adequate for this prototype level test, but the experiments have been designed for -10°C and testing at this level will give added confidence on the low temperature capability of the payload. The-10°C test has been requested by the Program Manager.

/s/ A. R. Timmins
A. R. Timmins, Head
Thermal-Vacuum Tests Section
Thermodynamics Branch

Enclosure

ART:gfr

cc: Office of Technical Services (1)
Thermodynamics Branch (10)
Thermal-Vacuum Tests Section (4)
Thermal-Vacuum Facilities Section (4)
Electronics Test Branch (2)
System Evaluation Branch (40)
Thermal Systems Branch (2)

THERMAL-VACUUM TEST PROCEDURE For S-3 PROTOTYPE SYSTEM PAYLOAD

FORWARD:

The test procedure outlined below is based on the latest calculated temperatures of the subject payload during its orbital life of one year. The payload will have several aspect positions relative to the sun during its life. This means that heating of one side occurs while the opposite side is being cooled; and later in orbit life, the heated and cooled surfaces will be reversed. The test procedure attempts to cover the simultaneous heating-cooling type of environment as well as the conventional uniform hot or cold environment.

The test will consist of four parts: (1) 45° Aspect, (2) 135° Aspect, (3) Hot, and (4) Cold. Test 1 attempts to simulate the space environment where the sun is at an angle of 45° to the longitudinal axis of the payload and illuminates the top side of the octagon. Test 2 attempts to simulate the space environment where the sun is at an angle of 135° to the longitudinal axis of the payload (bottom side of octagon illuminated). Figure D2-1 illustrates these angles and also shows various stations which are referred to in the test procedure. These stations correspond to locations for which temperatures have been calculated by the Thermal Systems Branch. Tests 3 and 4 are conventional tests in which the entire payload is soaked at a uniform temperature.

The external power supply is to be such that:

a. Payload can be operated solely on external power.

b. Payload can be operated solely on battery power.

c. Batteries can be charged during the test using the zener diode circuit.

TEST PROCEDURE FOR PROTOTYPE:

1. Preliminary Checkout

After all wiring has been completed and the payload is in the thermal-vacuum chamber, a complete system checkout will be made prior to starting the test. The chamber pressure will then be reduced with the payload operating and with no heating or cooling of the chamber. During this time the payload monitoring should be observed for any indication of arc-over. After the chamber pressure has reached a level of 1×10^{-4} mm Hg the payload power will be turned off. The chamber will continue to be evacuated to a pressure of 1×10^{-5} mm Hg (or better).

2. <u>45° Aspect</u>:

2.1 With payload power off, chamber wall temperature is lowered until the temperature of the transmitter (station 21) is stabilized at -20°C. (No station is to get to a temperature lower than -20°C). At the same time chamber wall is being cooled, light ring No.1 is adjusted to attain a stabilized temperature of 45°C at station six. Adequate instrumentation will be used to insure no hot spots on skin of octagon.

2.2 Light ring No.2 is off. The single light at the Magnetometer is adjusted to give a temperature of 30°C on the Magnetometer Box.

2.3 When thermal and pressure conditions are established, power (external) is turned on until the internal temperature (as indicated by the internal temperature of the main encoder) reaches equilibrium. (Temperature changes less than 0.5° per hour.)

2.4 Power is changed from external to battery operation for 30 minutes. Payload operation is checked.

2.5 Charge batteries using external power supply and the zener diode circuit. Continue until batteries are charged and station 16 (base of zener diodes) reaches temperature equilibrium.

2.6 Make a complete system check-out.

2.7 Turn off the single light on the magnetometer.

2.8 Estimated time for this test is 24 hours. However, the equilibrium conditions can be extended, if desired, so that the complete system check-out can be made at a convenient time.

3. <u>135° Aspect</u>:

3.1 With payload power off and with vacuum maintained from test 1, the chamber wall temperature is lowered until the temperature at station four is stabilized at -10°C.

3.2 Light ring No. 1 and the single light at the Magnetometer are off. At the same time light ring No. 2 is adjusted to attain a stabilized temperature of 35°C at station 12.

3.3 When thermal and pressure conditions are established, power (external) is turned on until the internal temperature (as indicated by the internal temperature of the main encoder) reaches equilibrium. (Temperature changes less than 0.5°C per hour.)

3.4 Power is changed from external to battery operation for 30 minutes. Payload operation is checked.

3.5 Charge batteries using external power supply and the zener diode circuit. Continue on external power after batteries are charged until temperature at station 16 (structure adjacent to the diodes) reaches equilibrium. (Temperature changes less than 0.5°C per hour.)

3.6 Turn on magnetometer light and determine operability as temperature is raised to 30°C. Turn off Magnetometer light.

3.7 Make a complete system check-out.

3.8 Estimated time for this test is 24 hours. However, the equilibrium conditions can be extended, if desired, so that the complete system check-out can be made at a convenient time.

4. <u>Hot Test</u>:

4.1 With payload inoperative and with vacuum maintained from test 2, the chamber wall temperature is adjusted to establish a uniform payload temperature of 35°C.

4.2 All lights in chamber are off.

4.3 With thermal and pressure conditions established, external power is turned on until the internal temperature (as indicated by the internal temperature of the main encoder) reaches equilibrium. (Temperature changes less than 0.5°C per hour.)

4.4 Power is changed from external to battery operation for 30 minutes. Payload operation is checked.

4.5 Charge batteries using external power supply operating through the zener diodes. Continue on external power after batteries are charged until temperature at station 16 (structure adjacent to the diodes) reaches equilibrium. (Temperature changes less than 0.5°C per hour.)

4.6 Test is continued until the cumulative time under vacuum conditions for tests 1, 2, and 3 is seven days, including the complete system check which is made at the end of test three. Payload is operated continuously during the hot test with continuous monitoring and a complete system check-out once each day.

5. Cold Test:

5.1 With payload power off and with vacuum maintained from test 3, the chamber wall temperature is adjusted to establish a uniform payload temperature of 0°C.

5.2 All lights in the chamber are off.

5.3 With thermal and pressure conditions established, external power is turned on until the internal temperature (as indicated by the internal temperature of the main encoder) reaches equilibrium. (Temperature changes less than 0.5°C per hour.)

5.4 Power is changed from external to battery operation for 30 minutes. Payload operation is checked.

5.5 Charge batteries using external power supply operating through the zener diodes.

5.6 Repeat paragraphs 5.1 through 5.5 except chamber wall temperature is maintained at a level to give -10°C in the payload (internal temperature of main encoder).

5.7 Test (at 0°C for payload) is resumed and is continued for three days including time for a complete system check-out at the end of the test. Payload is monitored continuously and a complete system check-out made once each day.

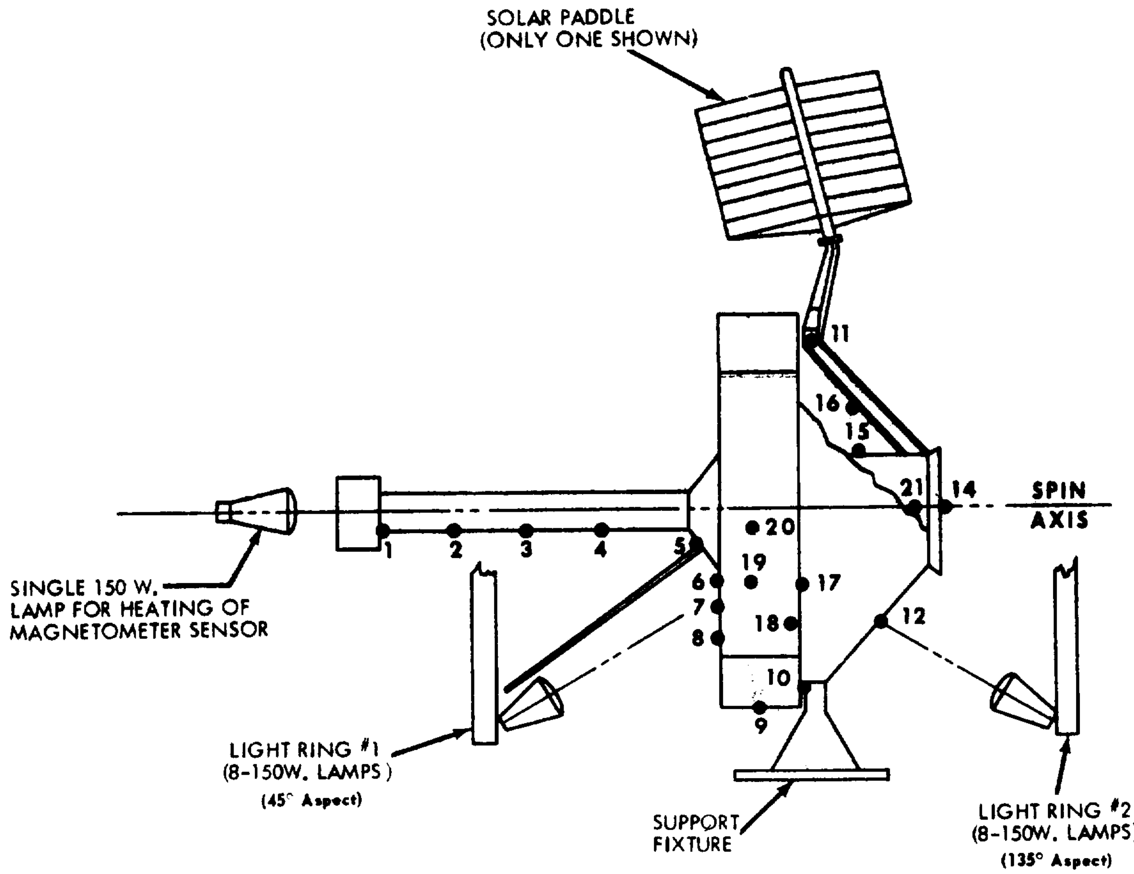
6. <u>Solar Paddle</u>:

6.1 A single Solar Paddle whose performance has been measured at the Naval Research Laboratory will be attached to the payload. Provision shall be made to illuminate each side of paddle with a 150 watt flood lamp.

6.2 During Test 1 (45°Aspect), after the paddle temperature had stabilized, the paddle will be illuminated on one side and the temperature of each side plotted until the temperature differential becomes constant. Turn off lamp. Repeat test, illuminating opposite side. In no case shall the surface temperature of the paddle be permitted to exceed 35°C.

6.3 At least once each day the voltage and current output of the Solar Paddle with both sides illuminated shall be recorded.

6.4 After completion of the thermal vacuum test the Solar Paddles shall be examined visually for any physical damage such as chips, cracks, of adhesive failures. The paddle also shall be returned to the Naval Research Laboratory for a performance test to determine any effects from the thermal vacuum test.



NOTE: Numbers refer to stations for which temperatures have been calculated by the Thermal Systems Branch. Skin temperature locations are indicated by dots attached to a line.

Figure D2-1 Setup For The Thermal-Vacuum Test Of The Prototype System Payload

APPENDIX E

ACCEPTANCE TEST PROGRAM

FOR

FLIGHT SYSTEMS

OF THE

S-3 ENERGETIC PARTICLES SATELLITE

326.3(FC)S-3-05

326.3(FC)S-3-05

ACCEPTANCE TEST PROGRAM
FOR
FLIGHT SYSTEMS
OF THE
S-3 ENERGETIC PARTICLES SATELLITE

PREPARED
BY

SYSTEM EVALUATION BRANCH
TEST AND EVALUATION DIVISION
OFFICE OF TECHNICAL SERVICES

MAY 1961

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

E-1

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ACCEPTANCE TEST PROGRAM

FOR

FLIGHT SYSTEMS

OF THE

S-3 ENERGETIC PARTICLES SATELLITE

PREPARED BY:

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MAY 23, 1961

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1.0 SCOPE

This document prescribes the Acceptance Test Program for the S-3, Energetic Particles Satellite. Environmental testing of the S-3 Flight Unit and Flight Spare will be done in accordance with provisions of this document.

All other documents relating to the Environmental Testing of S-3 Flight Units are obsolete and are hereby superseded.

2.0 INTRODUCTION

The S-3 Satellite will be launched by a Delta vehicle during the third quarter of 1961. The orbit will range from 150 nautical miles to a minimum apogee of 40,000 nautical miles. This orbit will be particularly advantageous for the study of the physics of fields and particles beneath, within, and above the Van-Allen radiation belts.

The Acceptance Test Program for the Flight System will consist of a dynamic balancing operation, a series of vibration exposures, and thermal-vacuum tests. In addition, a final dynamic balance will be done on the Flight Unit immediately prior to departure for AMR.

The sequence of testing shall correspond with the sequence of the following paragraphs of this document.

3.0 SYSTEM CHECKOUTS

The Flight System shall receive a complete operational checkout before balance, before and after vibration, before and after thermal-vacuum, and at such other times during the thermal-vacuum exposures as specified in Part 6 of this document.

3.1 Checkout Details

The following items shall apply for each complete system checkout:

3.1.1 Excitation of Sensors

A Cobalt-60 source shall be used to excite the Double Scintillation Telescope, the GM Telescope, the GM Omni Counter, and the SUI Electron Spectrometer.

Each of the three SUI-CDS sensors will be excited by an incandescent light bulb.

The Optical Aspect will be checked by using a light source containing six incandescent filaments (one for each of the six Optical Aspect Photo-diodes).

The Ion-Electron Detector will have simulated current and count inputs fed to it from its test box.

The Ames Proton Analyzer will be checked with the Ames Test Box which simulates the electrometer input currents.

The Magnetometer sensor shall be wrapped with external coils of wire so that by passing a DC current through these windings, the earth's magnetic field will be partially cancelled, and the resultant field strength will be within the range of the sensors.

A comprehensive check of Magnetometer calibration will be done in a controlled magnetic field, after the completion of the Acceptance Test Program.

The Solar-Cell Damage Experiment will be excited by an incandescent bulb. The voltage applied to the bulb shall be DC. The bulb shall be energized for a minimum amount of time, in order to prevent severe heating of the Damage Experiment.

3.1.2 Payload Operation

The payload shall be operated using external power supplied at the payload solar-paddle array interface.

The voltages and currents shall be recorded. The RF power output and frequency shall be measured and recorded.

Each of the 16 telemetry channels shall be monitored and recorded to verify correct operation of each sensor, associated subsystems, and telemetry sub-systems. Each subsystem will receive additional checkout, when applicable, to assure satisfactory operation.

The under-voltage "lock-out point" shall be measured and recorded. The operability of the Command Program Switch and the Recycle Timer shall be demonstrated, using the "fast" recycle time period.

The solar-celled paddles will be visually examined with a traveling microscope, tested using a standard light source, and certified to be Flight Acceptable before balance, after vibration, and after thermal-vacuum.

A detailed and accurate log shall be maintained by the Systems Integration Branch for each Flight System and should include all events concerning the system, its operation, environment, etc.

4.0 DYNAMIC BALANCE

The Flight Systems will be balanced on GSFC's Trebel Balancing Machine located in Building 4.

The balancing operation will be conducted at 150 RPM with the four live solar-celled paddles in the folded position.

The Flight System shall be in its exact flight configuration, with the following exceptions:

a. The Despin Timer weights shall be locked in place to prevent actuation of the despin device.

b. Non-strategic screws on the covers, etc. need not be secured with "Loc-Tite" until after balancing is completed.

In lieu of S-3 balancing restraints, the residual unbalance shall be less than the Douglas Aircraft specification for Delta payloads. The DAC specification allows 6.5 ounce-inches static unbalance and 480 ounce-inches-squared dynamic unbalance.

Such physical parameters as may be required to analytically compute the unbalance due to extended solar-celled paddles shall be recorded.

5.0 VIBRATION

5.1 Pre-Vibration Requirements

Prior to the initiation of vibration exposures, the system shall be checked-out in accordance with S-3 operation and checkout procedures. No vibration tests shall be conducted if the system exhibits abnormal operation or is not in its specified flight configuration.

The Cobalt-60 radioactive source shall be attached to the SUI GM Counter during each vibration exposure.

A RF link from the Flight System directly to the Ground Station Equipment shall be established. The Ground Station Equipment shall have the capability of recording the detected telemetry signal on magnetic tape.

There will be four solar-celled paddles mounted on the payload. Care shall be taken to assure that the specified amount of tension is in the chord used to secure the paddles in their folded position.

The vibration exposures will be conducted using GSFC's MB Vibration System.

5.2 Vibration Parameters

The vibration exposures shall be conducted in the following order:

a. Thrust Axis: sinusoidal, swept frequency

b. Thrust Axis: random

c. Thrust Axis: 600 cps, resonance

d. Transverse Axis (I-III, see Fig. 1): sinusoidal, swept frequency

e. Transverse Axis (I-III): random

f. Transverse Axis (I-III): 600 cps, resonance

g. Transverse Axis (II-IV, see Fig. 1): sinusoidal, swept frequency

h. Transverse Axis (II-IV): random

i. Transverse Axis (II-IV): 600 cps, resonance

Three calibrated accelerometers shall be mounted on the fixture as near to the fixture-payload interface as possible. Their sensitive axes shall be oriented so that one is trued with the axis of applied vibration and the other two are mutually perpendicular with the first.

Permanent records of the accelerometer signals shall be made and retained.

5.2.1 Sinusoidal Vibration

Frequency Range (cps)	Vector Acceleration (g) Thrust Axis	Vector Acceleration (g) Transverse Axes
5-50	1.5	0.6 (a)
50-500	7.1	1.4
500-2000	14	2.8
2000-3000	36	11.3
3000-5000	14	11.3 (b)

NOTES

(a) Within maximum amplitude limit of vibration generator.

(b) Within maximum frequency limit of vibration generator.

(c) The sweep rate shall be four octaves per minute.

(d) The duration of the exposure shall be 2-1/2 minutes each direction (total time - 7-1/2 minutes).

5.2.2 Random Vibration

Levels of exposure shall be in accordance with the following table for each of the three major axes:

Frequency Range (cps)	Spectral Density (g^2/cps)	Amplitude $g = \sqrt{g^2/cps \Delta f}$ (g-rms)	(a) Duration Min.
20-2000	.03	7.7	2.0

NOTES

(a) Two minutes each axis - total time: 6 minutes.

(b) White Gaussian noise with g-peaks clipped at three times the rms acceleration.

5.2.3 600 cps X-248 Resonance Vibration

Levels of vibration shall be in accordance with the following table:

Axis	Frequency	Force	Duration
Thrust	550-650 cps	±400 pounds (a)	15 seconds
Transverse Axes	550-650 cps	±67 pounds (c)	15 seconds

NOTES

(a) If it is not possible to program force with available equipment, the vector acceleration shall be determined by dividing ±400 lbs. (thrust axis) or ±50 lbs. (transverse axes) by the apparent weight of the payload, as measured over the 550-650 cps range.

(b) Each test is conducted by sweeping once at a rate so that 15 seconds are required to traverse the band from 550 cps to 650 cps.

(c) If the apparent weight of the payload must be assumed in order to conduct the transverse axes tests, in no case will an apparent weight of less than seven pounds be selected.

5.3 Payload Operation During Vibration Testing

The payload shall be operated on internal power (no charging of batteries) during each of the vibration exposures. The elapsed time on internal batteries shall be recorded. External power (to charge the batteries) shall be used immediately after each vibration exposure.

In addition to recording the detected telemetry signal during exposures, a 10-minute tape recording shall be made after tests a. through h. of paragraph 5.2.

System voltages, as measured by the Meter Panel, shall be continuously recorded during each vibration exposure.

Each of the solar-celled paddles will be visually examined and electrically checked to give a rough indication of their operability after each of the vibration exposures a. through h. (paragraph 5.2).

Antennae will be in the "down and locked" position for all vibration exposures.

At the conclusion of the entire series of vibration exposures, the Flight System shall be completely checked out in accordance with S-3 operation and checkout procedures.

6.0 THERMAL-VACUUM TESTS

6.1 Pre-Thermal-Vacuum Requirements

Prior to the initiation of the thermal-vacuum exposures, the system shall be checked out in accordance with S-3 operation and checkout procedures. This shall be accomplished after the payload has been installed in GSFC's Stokes 8'x8' Chamber and all instrumentation and set-up procedures have been completed.

The payload shall be in its exact flight configuration, except that the strippable coatings shall not be removed. This will facilitate handling and temperature control of the Flight System.

The payload monitoring equipment shall permit operation of the payload on internal power and on internal power with simultaneous charging of the batteries.

The batteries shall be fully charged.

Provisions shall be made so that sensors may be excited in accordance with paragraph 3.1.1.

Four live solar-celled paddles will be tested with the Flight System. Three paddles will be attached to the payload; the fourth will be secured to a fixture inside the chamber. (It is not possible to attach the fourth to the payload, since the fourth mounting strut is utilized to support the payload in the chamber.) Provisions shall be made to illuminate each side of each paddle.

6.2 Chamber Evacuation

The chamber pressure will be reduced to 1×10^{-4} mm Hg with no heating or cooling of the chamber. During this time, the payload shall be monitored continuously.

When the chamber pressure has reached 1×10^{-4} mm Hg, the payload will be turned off. The pressure will then be reduced to 1×10^{-5} mm Hg (or better).

6.3 30° Aspect Test

6.3.1 Thermal Conditions

a. With the payload power off, the chamber wall temperature shall be lowered until the temperature of the Transmitter (station 21, Figure 2) is stabilized at -20°C. Temperature stabilization is achieved when the temperature changes less than 0.5°C per hour.

b. While the chamber wall is being cooled, Light Ring #1 (Figure 2) shall be adjusted so that stations in the octagon do not achieve temperatures lower than -10°C.

c. After the stabilization of station 21 at -20°C is achieved, the payload shall be operated (internal power with simultaneous charging of the batteries). All payload voltages and currents shall be recorded, and a ten-minute tape recording of the detected telemetry signal shall be made.

d. With the payload power off, Light Ring #1 shall be adjusted to attain a stabilized temperature of +32°C on the Battery Pack B. The single light at the Magnetometer shall be adjusted to give a temperature of +35°C on the Magnetometer Sensor.

e. When thermal conditions have reached equilibrium, the payload shall be operated on internal power with simultaneous charging of the batteries, until the temperature of the Battery Pack B stabilizes.

6.3.2 Payload Operation

a. Light Ring #1 shall be turned off. Payload operation shall be changed to internal power with no charging of the batteries for 30 minutes. Payload voltages and currents and the transmitter frequency and output power shall be recorded. A ten-minute tape recording of the detected Telemetry signal shall be made and analyzed to assure correct operation of the system.

b. Light Ring #1 shall be turned on. The operation of the payload on internal power shall be continued, but with simultaneous charging of the batteries.

For each twelve hours of payload operation at equilibrium conditions, at least one complete system checkout shall be performed.

6.3.3 Test Duration

The test will terminate 48 hours after the payload is turned on, in accordance with paragraph 6.3.1-e. Time spent at equilibrium conditions may be extended, if desired, so that specific portions of the above procedure can be accomplished at a convenient time.

6.4 150° Aspect Test

6.4.1 Thermal Conditions

a. With the payload power off, the chamber wall temperatue shall be adjusted until the temperature at station #1 (Figure 2) stabilizes at -10°C.

b. At the same time, Light Ring #2 shall be adjusted to attain a stabilized temperature of +35°C at station 21. All other lights will be off.

c. When all thermal and pressure conditions are stabilized, the payload shall be operated on internal power (with simultaneous charging of batteries) until the temperature of the Battery Pack B stabilizes.

6.4.2 Payload Operation

a. Light Ring #2 shall be turned off. Payload operation shall be changed to internal power with no charging of the batteries for 30 minutes. Payload voltages and currents and the transmitter frequency and output power shall be recorded. A ten-minute tape recording of the detected Telemetry signal shall be made and analyzed to assure correct operation of the system.

b. Light Ring #2 shall be turned on. The operation of the payload shall be continued on internal power, but with simultaneous charging of the batteries.

For each twelve hours of payload operation at equilibrium conditions, at least one complete system checkout shall be performed.

6.4.3 Test Duration

The test will terminate 48 hours after the payload is turned on, in accordance with paragraph 6.4.1-c.

Time spent at equilibrium conditions may be extended, if desired, so that specific portions of the above procedure can be accomplished at a convenient time.

6.5 Cold Soak Test

6.5.1 Thermal Conditions

a. With vacuum conditions maintained from the previous tests, the chamber temperature shall be adjusted to establish a uniform payload temperature of -10°C.

The payload shall be operated (internal power with charging of the batteries) for the purpose of obtaining transmitter frequency and power output versus temperature information for the range of +35°C to -10°C. When the temperature of the transmitter reaches -10°C, the payload shall be turned off, and the stabilization of a uniform temperature (-10°C) shall be effected.

b. When stabilization of payload temperature has been achieved, the payload shall be operated using internal power with simultaneous charging of the batteries.

The conditions of this paragraph shall be maintained until the cumulative time of payload operation during paragraph 6.5.1 (a. and b.) is at least 24 hours. Two complete system checkouts shall be done during this 24-hour period.

6.5.2 Payload Operation

a. Payload operation shall be continued on internal power, but with no charging of the batteries. The time until payload "lock-out" shall be recorded.

During the 24-hour recycle time period, the batteries shall be charged using the external power supply.

When the payload is turned on at the end of the 24-hour recycle period, a complete systems checkout shall be made. During this system check, charging of the batteries shall be continued.

b. The temperature of the chamber walls shall be increased so that the internal temperatures of the payload reach +30°C within four hours. The payload shall be operated on internal power with charging of the batteries during this procedure, for the prupose of obtaining transmitter frequency and output power versus temperature information. When the temperature of the transmitter reaches +30°C, a complete system checkout shall be done. The chamber shall then be vented.

c. Time spent at equilibrium conditions during this cold-soak test may be extended, if desired, so that specific portions of this test can be accomplished at a convenient time.

S-3 ENERGETIC PARTICLES SATELLITE

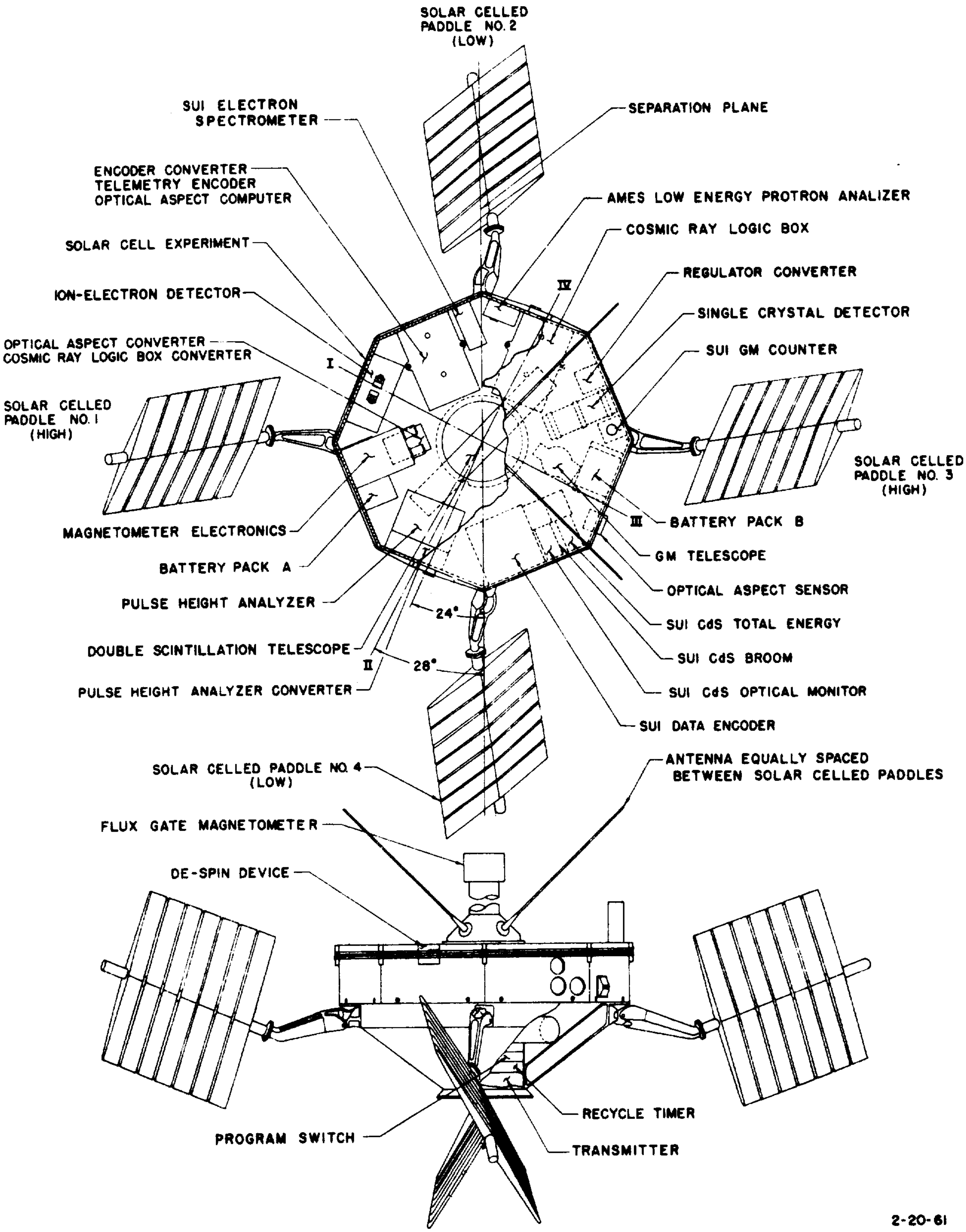


Figure 1.

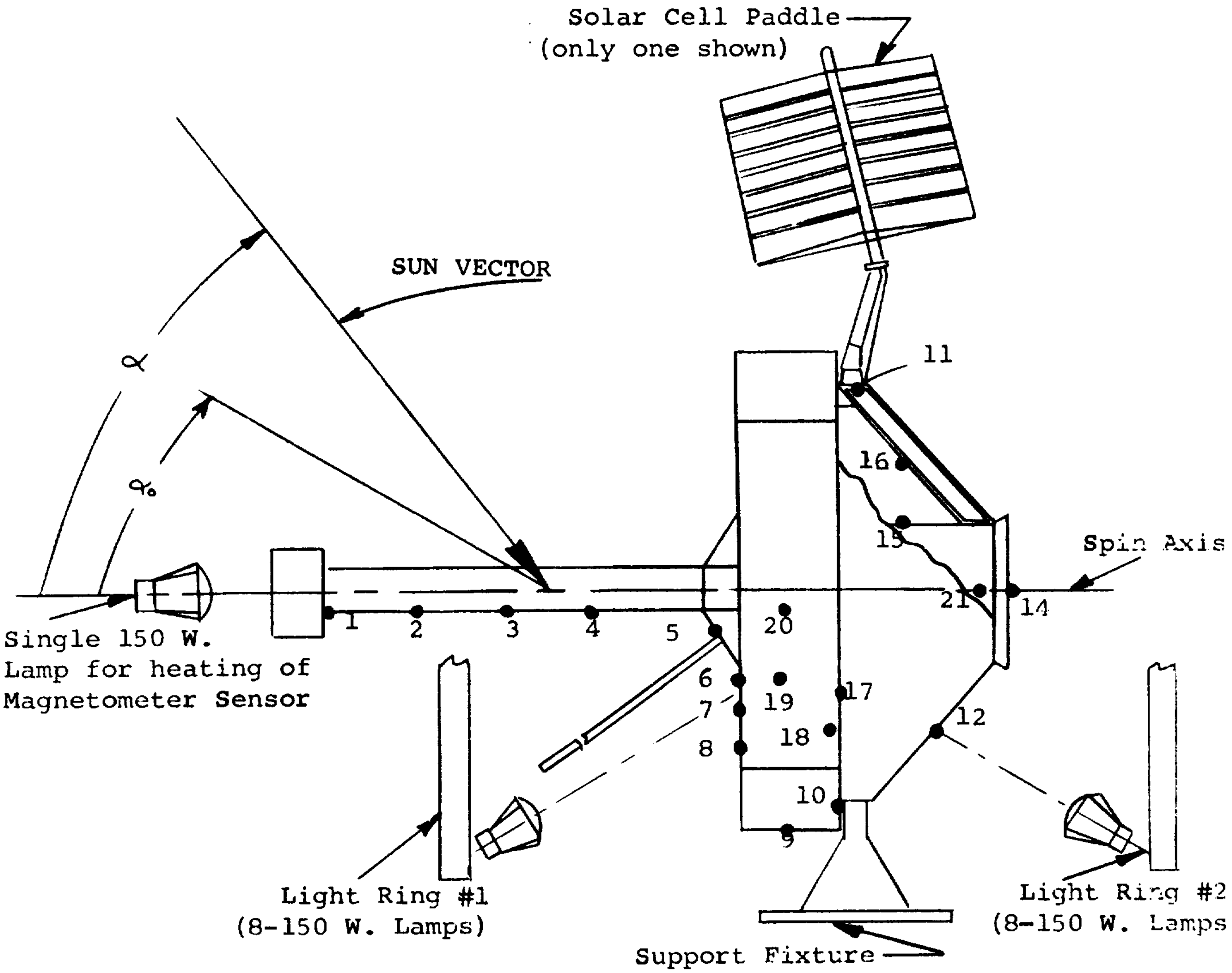
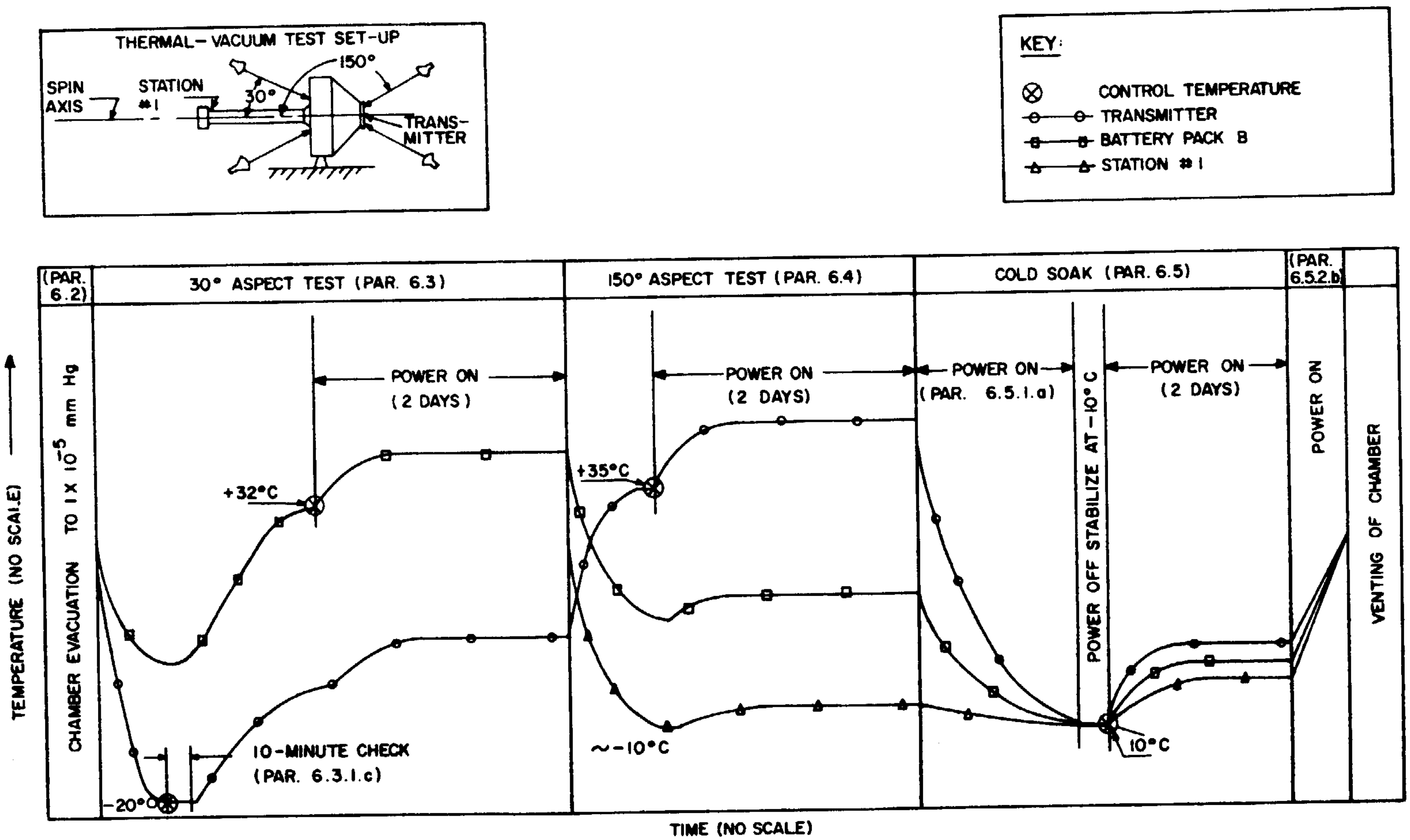


FIGURE 2

SET-UP FOR THERMAL-VACUUM ASPECT TESTS

NOTE: Numbers refer to stations for which temperatures have been calculated by the Thermal Systems Branch. Skin temperatures are indicated as dots attached to a line.

Angles referred to in the text are measured from the spin axis as shown.



ADDENDUM TO "ACCEPTANCE TEST PROGRAM FOR FLIGHT SYSTEMS OF THE S-3 ENERGETIC PARTICLES SATELLITE"
MAY 1961

Figure 3. Thermal-Vacuum Tests for Flight Systems of the S-3 Energetic Particles Satellite

APPENDIX F

TEST REPORT

STRUCTURAL PROTOTYPE

1.0 SUMMARY

The Structural Prototype "A" was exposed to vibration (October 8, 1960) per Test Plan dated October 3, 1960. The fracture, during vibration, of a stainless steel stud used to secure a simulated experiment to the instrument shelf necessitated repairs. The method of securing the item was modified by increasing the size of existing studs and adding two studs.

A major failure occurred when a small package (the Encoder Converter) tore loose from the honeycomb instrument shelf.

Both of these failures occurred during vibration in the 50 to 70 cps resonant frequency range of the payload in the axial (thrust) direction.

The latter failure required extensive repair of the payload. No further tests were conducted at this time.

The Structural Prototype (designated Model "B" because of modifications) was vibrated on November 3, 1960, per Test Plan dated October 24, 1960. Several minor failures (screws loose, slight separation of the skin from the honeycomb, gussets at the base of Magnetometer buckled slightly) were experienced, but none were judged to require extensive modification or repair of the structure.

Structural Prototype "B" was also exposed to the environments of sustained acceleration and spin. No failures or damage was observed as a result of these tests.

2.0 TEST RESULTS

2.1 Structural Prototype "A". -

2.1.1 Balance. - A static balancing operation was performed on the model with the following results:

Residual Unbalance ....$\leq$5.7 oz.-in.
Initial Unbalance .... 121 oz.-in.
Weight Added .... 0.59 lbs.

2.1.2 Vibration. - The transmissibility test indicated amplifications of 20:1 to 30:1 at several locations on the instrument shelf and on simulated experiments. These Q values were experienced with an input to the payload of ±1 g. The primary resonant frequency of the payload was about 80 cps. A gradual build-up from 50 cps to the maximum peak at 80 cps was observed.

A failure occurred in the 50-500 cps test range (simulated Encoder Converter tore loose from instrument shelf) and prevented further vibration in the thrust axis. Also, random, combustion resonance, and transverse axes vibration, acceleration, and spin were not attempted.

2.2 Structural Prototype "B". - The rebuilt prototype incorporated the following changes:

a. A new honeycomb instrument shelf of identical construction replaced the damaged original one.

b. An aluminum ring was added to help prevent the diaphragming action of the instrument shelf. The ring encircled the shelf, supporting it from underneath.

c. Four straps simulating the top cover, extending from the magnetometer column, over the instrument compartment, to the outer ledge of the unit, were added.

d. Weights were added at the base of the magnetometer column to simulate the antennas.

2.2.1 Balance. - A static balancing operation was performed on the model with the following results:

Residual Unbalance .... Less than 19.2 oz.-in. but greater than 12.5 oz.-in.

Initial Unbalance ..... 94.5 oz.-in.
Weight Added .... 0.42 lbs.

2.2.2 Vibration in thrust direction. - Vibration began on November 3, 1960, in accordance with the Test Plan for Structural Prototype (October 24, 1960). *

* See Appendix C

2.2.2.1 The probing survey at an input of ±1 g indicated high values of acceleration at some locations. Regions of high amplifications were located using a hand-held probe and calibrated Endevco accelerometers were secured at these locations. During the course of this survey it was noticed that the GM Telescope was not fastened properly to the instrument shelf. At least one of the fasteners was oversized so that there was a space between the package and the shelf.

This condition may have caused a weakening of the adhesive bond between the honeycomb and the facing during vibration. A separation of the two was first noticed in the vicinity of the telescope after the transmissibility (1 g) survey. Nothing was done at this time to rebond the facing to the honeycomb.

The fastener difficulty was overcome by a few thin metal shims.

2.2.2.2 The "Flight Acceptance" levels sinusoidal vibration resulted in no failures. The structure resonated from 100 to 130 cps. Amplifications of 11:1 (±7 g input) were recorded by accelerometers on the instrument shelf. The magnetometer column resonated at 210 cps and, also, in the 400-450 cps range. Amplifications of 14:1 (±7.0 g input) were recorded on the Magnetometer in both frequency ranges.

2.2.2.3 After the "Qualification Test" levels - sinusoidal vibration, the payload was inspected, and the following observations were made:

a. Additional separation of the facing and the core of the honeycomb instrument platform in the vicinity of the GM Telescope had occurred.

b. Four of the screws used to attach the instrument platform to the center tube had backed out. Two more had sheared.

c. The screws holding the magnetometer column to its base had come out.

d. The rings simulating the antenna had come loose.

These conditions necessitated repairs and a repeat of the above exposure.

2.2.2.4 A retest (after repairs of all but the honeycomb-facing separation condition) at Design Qualification levels from 50 cps to 3000 cps caused no apparent failures. The accelerometers recorded the following amplifications with a ±10.3 g input:

Q = 8 at the bottom of honeycomb, approximately 100 cps.

Q = 10 on the GM telescope, approximately 100 cps.

Q = 14 on the top of the Magnetometer approximately 440 cps.

2.2.2.5 Random vibration produced no serious failures. Only a single screw holding one of the straps (which simulated the top cover) came loose.

2.2.2.6 The combustion resonance (600 cps sinusoidal vibration) test was conducted by assuming an apparent weight of the payload of five pounds and exposing the payload to the maximum vibrator output in the 550 to 650 cps range. This output was measured to be ±40 g which is considerably lower than the 86 g-rms specified for a payload having an apparent weight of five pounds. The magnetometer column experienced a Q of 2. At all other data points, the vibration was attenuated by the structure in this frequency range.

After this exposure, the vibration schedule was halted to permit detailed examination of the payload. It was discovered that the screws used to attach the simulated Magnetometer sensor to the top of the column were loose.

The honeycomb in the area of the GM Telescope was also repaired at this time by the injection of a bonding material. The facing had separated from the honeycomb in this area at the very beginning of the vibration testing because of an improper fastener. Once weakened, the honeycomb had separated an additional amount with each succeeding vibration exposure.

Also, modified at this time was the method of securing the GM Telescope to the shelf. A bracket was added attaching the Telescope to the Battery Pack B thus securing the Telescope package in two planes.

In light of the above examination and subsequent modifications, it was agreed that a retest of at least a part of the vibration schedule would yield profitable transmissibility data. This data would eliminate any question of large amplifications having been caused by loose screws, weakened honeycomb, prior methods of securing packages, etc..
Also, supporting the desirableness of vibration was the fact that accelerometer recording instrumentation difficulties and accelerometers breaking off from the unit due to high amplifications prevented the acquisition of complete transmissibility data during previous vibration exposures.

2.2.2.7 <u>± 1 g Input Retest</u>. - A single sweep at one octave per minute, from 5-3000 cps, was conducted at an input of ± 1 g in the thrust direction. The following maximum amplifications were recorded:

a. Under the instrument shelf
Q = 12 - 88 cps

b. Top of the Magnetometer
Q = 45 - 500 cps

c. Top of the GM Telescope
Q = 18 - 100 cps

d. On the octogon wall
Q = 20 - 88 cps

e. On the floor of the instrument shelf
Q = 15 - 100 cps

2.2.2.8 <u>"Flight Level" retest</u>. - The second retest was a sweep from 5-3000 cps, thrust axis, at "Flight Acceptance" levels. The sweep rate was four octaves per minute.

The accelerometers indicated the following peak readings:

a. Underneath the instrument shelf:
±90 g, ±7 g-input (Q=13), 90 cps

b. Magnetometer:
Vertical Direction:
±150 g, ± 7 g-input (Q=21) 450 cps
± 75 g, ±14 g-input (Q= 5) 720 cps

Horizontal Direction:
±40 g, 450 cps

c. GM Telescope:
±60 g, ±7 g-input (Q=8.5) 90 cps

d. Floor of the Instrument shelf:
±105 g, ±7 g-input, (Q=15), 90 cps

2.2.2.9 Magnetometer column resonance exposure. - Personnel of the Mechanical Systems Branch requested that another vibration exposure at lower levels be conducted in the resonant frequency range of the magnetometer column. The Test and Evaluation Division personnel suggested that a slow sweep rate (one octave per minute) from 300 to 600 cps (encompassing the entire frequency range--from build-up through peak amplification on the column) at one-half "Flight Acceptance" levels be performed. The exposure was conducted as follows:

Input .... ±3.5 g
Frequency Range...... 300 to 600 cps
Sweep Rate........... 1 octave per minute

The Magnetometer experienced a maximum acceleration of ±110 g in the thrust direction (Q=31, 410 cps) and ±12 g in the horizontal direction at 410 cps.

2.2.3 Vibration in transverse axes. -

NOTE: The transmissibility information presented in this appendix was taken from oscillograph recordings and was not spectrum analyzed for harmonic components in the waveform.

2.2.3.1 Axis 1. - The orientation of this axis of vibration is as follows: The axis passes through the center of the octagonal instrument shelf and bisects two opposite angles, neither of which is supported by a strut. This axis is parallel to the axis of the Double Scintillation Telescope.

2.2.3.1.1 The "Flight Acceptance" level test caused no observed failures. Amplifications recorded were less than 10:1 except on the Magnetometer which experienced an acceleration of nearly ±40 g (Q=66, input ±0.6 g, 30 cps).

Above 2000 cps the fixture started to resonate with an approximate (maximum) Q of 10 at 2500 cps. The payload was then tested with the control input at the top of the fixture, and much better control results were obtained. Accelerations of ±60 g and ±42 g were recorded on the GM Telescope and the Magnetometer (vertical direction), respectively.

2.2.3.1.2 Inspection of the payload after the test at "Qualification" levels revealed that the four flanges or gussets at the magnetometer column base had buckled slightly (estimated to be about 1/32 of an inch at the midpoint of the gusset.) No repairs were necessary.

The magnetometer column resonated at about 30 cps, Q=60 (±50 g with an input of ±0.85 g).

2.2.3.1.3 X-248 Resonant Burning. - The 600 cps combustion resonance was conducted by sweeping the frequency spectrum from 550 cps to 650 cps in 30 seconds. Again an apparent weight of five pounds was assumed, and the payload was exposed to the maximum obtainable output of the vibrator. Input to the payload was measured to be ±20 g which is approximately equal to the 15 g-rms specified for a payload with an apparent weight of 5 pounds. No failures were observed.

Amplifications ranged from 2:1 to 4:1 on the GM telescope, on the wall of the instrument shelf, and in the horizontal and vertical directions on the Magnetometer. On the instrument shelf, amplifications were less than 1:1.

2.2.3.1.4 Random vibration. - The random vibration exposures were conducted in the 20 to 500 cps frequency range only. (See paragraph 2.3.d.):

Spectral Density ..... 0.12 g^2/cps
Amplitude .... 7.6 g-rms
Duration .... 4 minutes

The power spectral density used for control was 0.12 g^2/cps rather than the specified 0.1 g^2/cps. The difference between these is not unduly significant, since the PSD could have differed at any particular resonance by ± 0.05 g^2/cps and still remained within specification tolerances.

The magnetometer column (at the top) experienced large displacements throughout the entire exposure.

2.2.3.2 Axis 2. - The second axis chosen for transverse vibration was perpendicular to Axis 1. Flight and Qualification level sinusoidal vibration exposures were conducted, and no failures were observed. Amplifications recorded were nearly identical to those obtained for Axis 1. (See paragraphs 2.2.3.1.1 and 2.2.3.1.2.)

All conditions and remarks pertaining to the combustion resonance (600 cps) vibration and the random vibration of Axis 1 are equally applicable to Axis 2.

No failures were observed as a result of the combustion resonance test.

Slight additional buckling of the magnetometer flanges or gussets may have occurred during the random vibration exposure.

NOTE: Chronologically, all sinusoidal exposures in both transverse directions were completed before conducting the two random vibration exposures.

2.2.4 Acceleration exposure. - The structural prototype (with the paddle arms secured in the folded position by the vibration fixture) was mounted on the 20' radius (nominal) centrifuge of the Naval Research Laboratory at the Chesapeake Bay Annex.

A radial acceleration in the thrust direction of 28 g was maintained for one minute.

The acceleration of 28 g allows a 50% safety factor above the maximum expected powered flight acceleration.

No failures were detected as a result of this test.

2.2.5 Spin. - The structural integrity of the payload during spin was verified during an experimental dynamic balancing operation conducted in December 1960. A maximum spin rate of 180 rpm was used for the dynamic balance.

Due to the time required for a dynamic balance, the payload was spun at 180 rpm for a much longer time than the specified 1-½ minutes.

2.3 Deviations from Environmental Test Specifications - S-3 Energetic Particles Satellite. -

a. Vibration

(1) The highest frequency was machine limited to 3000 cps.
(2) The maximum g level was ±40 g due to vibrator limitations.

b. A vibration exposure specifying a three-minute dwell at each resonant frequency (which appeared in an earlier version of the Test Plan was omitted at the request of the Mechanical Systems Branch.

c. Several additional vibration retests were conducted because of previous failures. Also, an additional vibration sweep was conducted at the request of the Mechanical Systems Branch. (See paragraph 2.2.2.9.)

d. The random vibration in the transverse axes was limited to the 20-500 cps range. On this test, there was some difficulty in equalizing the fixture for a constant spectral density input. The test was attempted, but fixture resonances prevented conducting it throughout the entire spectrum. It was agreed since there were no serious structural resonances in the range from 500 to 2000 cps as evidenced by transmissibility data, it would be reasonable to limit the upper frequency for the random vibration test to 500 cps.

TABLE 1

Structural Prototype "B"
Thrust Axis — Sinusoidal Vibration

Sinusoidal Vibration Exposure Parameters	ACCELEROMETER LOCATIONS: Instrument Platform			Top of GM Telescope			Magnetometer (Vertical)			Magnetometer (Horizontal)			Under Instrument Platform		
	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q
1 g-vector, 5-3000 cps,															
1 oct./min. Sweep	88	12	12	100	18	18							100	15	15
							500	45	45						
							900	12	12						
							2800	12	12						
<u>Flight Levels</u>															
cps: 5-50 : 1.5 g-vector															
50-500 : 7g	90	90	13	90	60	8.5	90	25	3.5	90	35	5	90	105	15
				300	10	1.5	120	40	5.5	120	30	4	180	20	3
							450	150	21.0	420	40	5.5			
							500	40	5.5						
500-2000: 14g							550	62	4.5	580	20	1.5			
							600	40	3	630	20	1.5	900	67	5
	800	20	1.5	1000	10	1	720	75	5.5	1500	15	1			
							1820	50	3.5	1800	15	1			
							2000	40	3						
2000-3000: 40g				3000	10	1	2200	140	3.5						
4 oct./min. Sweep															
<u>Prototype Levels</u>															
cps: 50-500 : 10.3g															
2 oct./min. Sweep	100	81	8	100	100	10	440	148	14.5						
<u>Magnetometer Column Resonance Exposure</u>															
300-600 cps: 3.5 g-vector							300	10	3	300	Neg.	-			
1 oct./min. Sweep							350	20	6						
							400	40	11.5	400	10	3			
							410	110	31	410	10	3			
							500	20	6						
							600	10	3	600	Neg.	-			

Note: g_0 is the acceleration recorded at the indicated locations and assumes a sinusoidal waveform.

TABLE 2

Transverse Axes — Sinusoidal Vibration

Sinusoidal Vibration Exposure Parameters	ACCELEROMETER LOCATIONS														
	Instrument Platform Wall			Magnetometer (Vertical)			Magnetometer (Horizontal)			GM Telescope			Instrument Platform		
	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q	f cps	g_0 Vector	Q
Transverse Axis 1															
Flight Levels:															
5-50 cps, 0.6g							30	40	67						
50-500 cps, 1.5g				440	20	-									
500-2000 cps, 2.8g	680	22	8												
2000-3000 cps, 11g				2200	42	-				2200	60	5.5			
Prototype Levels:															
5-50 cps, 0.85g							30	50	59						
50-500 cps, 2.1g															
500-2000 cps, 4.3g	600	40	9												
2000-3000 cps, 16.9g										2200	80	4.5			
Transverse Axis 2															
Flight Levels:				30	12	-	30	35	58						
				450	25	-									
	600	15	5.5	700	22	-									
										1900	50	18	1900	15	5.5
Prototype Levels:				28	15	-	28	50	59						
	100	20	9.5												
				440	35	-									
	600	30	7				600	20	4.5						
				700	40	-									
							1300	20	4.5						
				1900	25	-				1900	55	13			

Note: g_0 is the acceleration recorded at the indicated locations and assumes a sinusoidal waveform.

APPENDIX G

PROBLEM AREAS ENCOUNTERED

DURING

ENVIRONMENTAL TESTING

OF THE

S-3 PROTOTYPE UNIT

TABLE I

PROBLEM AREAS ENCOUNTERED DURING ENVIRONMENTAL TESTING OF THE S-3 PROTOTYPE UNIT

JUNE 23, 1961

SYSTEM EVALUATION BRANCH
TEST AND EVALUATION DIVISION

	SUB ASSEMBLY	1 PRIOR TO BALANCING 22 MARCH 1961	2 SPIN 25 MARCH 1961	3 ACCELERATION 28 MARCH 1961	4 THRUST AXIS SINUSOIDAL 30 MARCH 1961 (VIBRATION)	5 REPEAT OF THRUST AXIS SINUSOIDAL 31 MARCH 1961 (VIBRATION)	6 RANDOM AND 600 cps THRUST AXIS 1 APRIL 1961 (VIBRATION)	7 TRANSVERSE AXES 3 APRIL 1961 (VIBRATION)	8 TEMPERATURE 4 APRIL 1961	9 FIELD TESTS 7 APRIL 1961	10 FIRST PRE-TEST SET-UP & CHECKOUT 14-17 APRIL 1961	11 FIRST ATTEMPT AT ASPECT TESTS 17-20 APRIL 1961 (THERMAL-VACUUM TESTING)	12 FIRST HOT (+35°C) SOAK 21-24 APRIL 1961 (THERMAL-VACUUM TESTING)	13 FIRST COLD (-10°C) SOAK 25 APRIL 1961 (THERMAL-VACUUM TESTING)	14 REWORK & SECOND PRE-TEST OPERATIONS 28 APRIL - 3 MAY (THERMAL-VACUUM TESTING)	15 (SECOND) HOT SOAK TEST (+35°C) 3-9 MAY 1961 (THERMAL-VACUUM TESTING)	16 (SECOND) COLD SOAK TEST (-10°C) 10-11 MAY 1961 (THERMAL-VACUUM TESTING)	17 REWORK & THIRD PRE-TEST OPERATIONS 12-18 MAY 1961 (THERMAL-VACUUM TESTING)	18 45°C ASPECT TEST 19 MAY 1961 (THERMAL-VACUUM TESTING)	19 135°C ASPECT TEST 22 MAY 1961 (THERMAL-VACUUM TESTING)	20 FINAL HOT (+35°C) CHECK 24 MAY 1961 (THERMAL-VACUUM TESTING)	21 SHOCK 31 MAY 1961	22 HUMIDITY 16 JUNE 1961	23 VIBRATION RETEST 24 JUNE 1961	24 VACUUM RETEST 25 JUNE 1961
A	TRANSMITTER				■ [R]									◧ [M]			[?]								
B	PROGRAM SWITCH											[?]		[?]		◧		[D]	[?]						
C	RECYCLE TIMER															⬓		[D]							
D	REGULATOR CONVERTER			[C]		[?]	[?]	[?]					◧												
E	CURRENT SENSOR	⊠							⊠ [C]		⊠ [M]	[C]	◫		[R]										
F	SOLAR ARRAY VOLTAGE REGULATOR	⊠									[D]		■		[D]										
G	BATTERIES														[M]										
H	GM TELESCOPE						◪ [M]											◪						◪	
I	SINGLE CRYSTAL DETECTOR												◪		[C]										
J	DOUBLE TELESCOPE																		◪						
K	PULSE HEIGHT ANALYZER									◪ [M]			◪		[M]		◪	[R]							
L	ION-ELECTRON DETECTOR				◪ [R]						◫														
M	MAGNETOMETER ELECTRONICS											[?]	◪		[M] ◫ [R]										
N	AMES PROTON ANALYZER						[?]		◫		◫														
O	SUI GM COUNTER	◪	[?]	[?] [M]											◪ [M]				◪ [C]						

IF THE SUBASSEMBLY MALFUNCTION (OR OTHER UNSATISFACTORY PERFORMANCE) OCCURRED IN FLIGHT, IT COULD HAVE CAUSED:

- ■ A MISSION FAILURE
- ◧ A PARTIAL MISSION FAILURE BUT ONLY WITH AN IMPROBABLE SIMULTANEOUS COMBINATION OF CIRCUMSTANCES
- ◪ A LOSS OF DATA FROM THE AFFECTED EXPERIMENT

OTHER PROBLEMS ENCOUNTERED ARE DENOTED BY:

- [?] QUESTIONABLE SUBASSEMBLY OPERATION.
- ⬓ MARGINAL SUBASSEMBLY OPERATION.
- ⊠ PROCEDURAL FAILURE.
- ◫ SPECIAL PROBLEMS.
- [R] SUBASSEMBLY REPAIRED.
- [C] SUBASSEMBLY CHANGED (REPLACED).
- [M] SUBASSEMBLY MODIFIED.
- [D] SUBASSEMBLY REDESIGNED.

S-3 PROTOTYPE UNIT TESTING SUMMARY

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
A-4	3/30/61	Transmitter	Current dropped from a normal 400 ma to about 150 ma: output power dropped from a normal 1-1/2 watts to about 0.1 watt — this is indicative of burnout of final stage.	Vibration - sinusoidal, thrust axis	Final tube found to be defective. It appeared to have a broken weld within the envelope. The tube was replaced.	Vibration test repeated, no recurrences.
A-13	4/25/61	Transmitter	Final stage refused to turn on at −12.5°C. Temperature raised: unit functioned normally at −7°C.	Thermal-Vacuum — first cold soak exposure	Trouble due to Transmitter Converter; starting resistor changed.	Thermal-vacuum exposure repeated.
A-16	5/9/61	Transmitter	Failed to turn on properly at −10°C.	Thermal-Vacuum — second cold soak exposure	Test halted 5/12/61. Subassembly testing of transmitter began 5/16/61. Starting characteristics at −20°C, −10°C, and 0°C normal. Twelve-hour soak at −20°C and $1X10^{-5}$ mm Hg indicated no deficiencies. Item replaced in Prototype Unit.	Suspicion that cables and/or loading may have caused problems of 5/9/61. Due to an oversight, the oscillator which had been detuned to aggravate the (subassembly test) cold-starting conditions had not been readjusted prior to reinstallation in the prototype system.
B-11	4/19/61	Program Switch	Unit seemed to respond very sluggishly at low temperatures.	45° Solar Aspect exposure (first attempt)	None	Possibility that the S-3 Blockhouse Control was not yet warmed up.

NOTE: The key refers to individual items in the preceding table.

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
B-13	4/25/61	Program Switch	The on-off switch of the S-3 Blockhouse Control does not turn the spacecraft off.	Thermal-Vacuum – first −10°C exposure	None	
B-15	5/7/61	Program Switch	Rapidly decreasing battery voltage fails to turn spacecraft off.	Thermal-Vacuum – second hot soak exposure	Card removed for rework 5/13/61. Additional amplification added to the "off" bus for under-voltage turn-off. Condenser storage added to "hold-up" under voltage circuit when Batteries collapse. Subsequent check (5/18/61) indicated perfect operation.	This problem was caused when the external charging supply was accidentally interrupted, thereby throwing the load to the batteries when the Battery voltage was just above the lock-out point.
B-17	5/13/61	Program Switch	See B-15 above		See B-15 above	
B-19	5/20/61	Program Switch & Recycle Timer	Blockhouse Control would not turn spacecraft on or off; Recycle Timers drawing slightly more current than usual. Repeated attempts to command the spacecraft unsuccessful.	Second 45° Solar Aspect exposure. Temperature of card: −11°C	None, test continued.	Undervoltage lock-out successful; spacecraft turned back on normally. Oscillator periods are .037 sec. and .058 sec. Later measurement was .041 sec. and .070 sec. Oscillator #1 stopped when payload turned on, but on 5/18/61 both oscillators continued running after spacecraft turned on.

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
C-15	5/8/61	Recycle Timer	Spacecraft was not turned on by fast oscillator. It was turned on by the (redundant) slow oscillator.	Second Hot Soak, Thermal-Vacuum	Card removed (5/13/61); pulse stretcher added to relieve marginal turn-on characteristics. Mis-adjustment of clock timer accounts for 27-hour and 11-hour periods. Clocks adjusted for 24-hour periods.	During $-10^{\circ}C$ cold test of 5/12, the fast oscillator did turn the spacecraft on.
C-17	5/13/61	Recycle Timer	See C-15 above		See C-15 above	
D-3	3/27/61	Regulator Converter	Poor regulation of 6.5 v supply. Problem had been recognized prior to balance operations.	Prior to Acceleration tests	"Flight Spare" converter installed.	
D-5	3/31/61	Regulator Converter	Output voltages from unit experienced a 60 ms transient drop-off.	Vibration — second sinusoidal thrust test; at approximately 85 cps, and about ±40g	None	
D-6	4/1/61	Regulator Converter	Voltage varied during test. (6.5 volt line)	Vibration — random thrust axis	None	Post-test checkout successful.
D-7	4/3/61	Regulator Converter	Voltage erratic during test. (6.5 volt line)	Vibration — transverse axes	None	Post-test checkout successful

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
D-12	4/21/61	Regulator Converter	Excessive temperature rise when internal power on.	Thermal-Vacuum — first hot soak exposure	Prior to the second hot soak exposure, all packages in the octagon were painted black; a conductive heat radiator added to the region of the conductor.	
E-1	3/16/61	Current Sensor	Unit has an open winding which prohibits charging of the Batteries through the Solar Array. Winding had been burned out due to incorrect solar regulator circuit.	Prior to Balance	None	Current Sensor is incorporated in the Battery Pack A in the Flight Spacecraft. See Item F-1.
E-8	4/5/61	Current Sensor	Operation of the spacecraft on internal power not possible.	Temperature test. −10°C operative soak	Test suspended; unit had open winding between the batteries and the turn-on plug. Apparently caused by a defective cable connected during pre-test set-up operations. Flight Unit #1 sensor installed.	
E-10	4/14/61	Current Sensor	Unit has an open winding between the Solar Array and the Batteries. Caused when leads were shorted when installing the Solar Array voltage regulator circuit.	Pre-Vacuum set-up operations	Pins were shorted so that batteries could be charged.	

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
E-11	4/18/61	Current Sensor	None except as noted in E-10.	Pre-Vacuum set-up operations before first hot soak exposure	Unit #3 installed.	
E-12	4/22/61	Current Sensor	Unit has open winding between Batteries and the simulated Solar Array. Winding apparently burned out because of malfunction of Solar Array voltage regulator.	Thermal-Vacuum — first hot soak exposure	None	
E-14	4/28/61	Current Sensor	None except as noted in E-12.	Set-up operations for second hot soak thermal-vacuum exposure	The unit was rebuilt and replaced in spacecraft.	
F-1	3/16/61	Solar Array Voltage Regulator	Incorrect Zener diodes had been installed for "mechanical simulation" only. These diodes were mistakenly connected. When the spacecraft was operated on internal power, the batteries discharged through the Zeners burning out the Current Sensor.	Pre-Balance checkout	None	
F-10	4/14/61	Solar Array Voltage Regulator	None except as noted in F-1	Pre-Vacuum set-up operations	Design change. Zener diodes removed and replaced by a transistor voltage regulator.	

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
F-12	4/21/61	Solar Array Voltage Regulator	Apparent thermal runaway of transistor caused burnout of Current Sensor and prohibited charging of Batteries.	Thermal-Vacuum — first hot soak exposure	None at this time.	
F-14	4/26/61	Solar Array Voltage Regulator	As noted in Item F-12		Circuit redesigned for improved heat dissipation. Now includes a parallel combination of two power transistors and two power resistors.	
G-14	4/26/61	Battery Pack	Prototype battery packs have one more cell than flight units (14 as compared to 13).		One cell shorted out of circuit.	
H-6	4/1/61	GM Telescope	No readout on post-thrust axis checkout. Loss of readout caused by the failure of one of the "pancake" geiger tubes	Vibration — sinusoidal, random, and 600 cps, thrust direction	Since no tubes were available for replacement, the coincidence readout was switched to the operative geiger tube.	
H-17	5/19/61	GM Telescope	No readout.	Pre-test checkout at standard temperature and pressure	None, test started.	A new "prototype" Telescope was successfully accelerated and vibrated (as a subassembly, 6/16/61), but subsequently failed during pump-down of a thermal-vacuum test.

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
H-23	6/24/61	GM Telescope	Unit not operating.	Equilization procedure for vibration shaker	None	
I-12	4/21/61	Single Crystal Detector	Encoder format incorrect. Turning the Double Telescope and Single Crystal Detector switch-off remedies situation. 72 hours later (after starting cold soak) interference with encoder disappeared.	Thermal-Vacuum – first hot soak exposure	Complete failure of high voltage power supply. HVPS had extensive arcing and produced corona during its death throes. During this time it caused interference with encoder. When the HVPS failed completely (72 hours later), no interference was produced.	During the 72-hour interval prior to complete failure of the power supply, the detector was actually on for only a short period of time.
I-14	4/28/61	Single Crystal Detector	As noted in Item I-12.		"Flight Spare" detector installed in payload.	Pre-test checkout 5/1/61 indicated high gain on spare SCD. Gain had not been previously adjusted on this unit. Adjustment was made and unit installed in payload 5/3/61.
J-18	5/21/61	Double Scintillation Telescope	Incorrect readout from the Pulse Height Analyzer	45°C Solar Aspect exposure	None, test continued.	Double Telescope was subsequently tested at −20°C in a vacuum, at which time it was determined that the power supply had malfunctioned.

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
K-9	4/17/61	Pulse Height Analyzer	Binary noise indicated in readout. Extremely high data acquisition with no input.	After Temperature testing during RF field checks	Transistor replaced in reset circuit. May have been marginal. Cold solder joint discovered in the sensitivity trigger. A jumper was added.	
K-12	4/21/61	Pulse Height Analyzer	Loss of readout.	Thermal-Vacuum — first hot soak (+35°C)	Problem traced to a cold solder joint. A wire was connected from the terminal post with the cold solder joint to the next terminal post.	
K-14	4/26/61	Pulse Height Analyzer	As noted in Item K-12		See Item K-12 above.	
K-16	5/11/61	Pulse Height Analyzer	Loss of readout	Thermal-Vacuum — second cold soak at −10°C		
K-17	5/12/61	Pulse Height Analyzer	As noted above, Item K-16		Transistor in decoding gate was temperature sensitive; transistor changed.	
L-4	3/30/61	Ion-Electron Detector	Incorrect readout.	Vibration — sinusoidal, thrust direction	Photomultiplier malfunctioned. Tube replaced.	Test repeated, no recurrences.

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
L-10	4/14/61	Ion-Electron	Electrometer fails to respond to excitation.	Pre-vacuum set-up operations	The I&E unit was o.k.; however, the test cable used to excite the experiment was defective. Cable connector repaired — operation normal.	Instrumentation difficulty.
M-11	4/18/61	Magnetometer Electronics	Electronics calibrated five times in rapid succession. Normal operation thereafter.	Chamber evacuation	None	
M-12	4/22/61	Magnetometer Electronics	Calibration circuit drawing excess current. Malfunction continued for major portion of hot soak exposure.	Thermal-Vacuum — first hot soak exposure	None	Unit functioned properly in subsequent cold soak test.
M-14	4/26/61	Magnetometer Electronics	As noted above		Unit returned to Univ. of N.H. Failure assumed due to reverse voltage applied. Two transistors which were causing the trouble were replaced. The bias resistors on these amplifier transistors were slightly changed to further decrease possibility of thermal runaway.	The unit was reinstalled in the prototype system, but was again malfunctioning. Apparently in the process of potting after repairs, the circuit was disrupted. The unit was depotted, fixed and potted again.
N-6	4/1/61	Ames Proton Analyzer	The current drawn by the Ames unit was erratic during vibration. Post-test checkout satisfactory.	Thrust axis, random vibration	None	

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
N-8	4/6/61	Ames Proton Analyzer	Ames readout incorrect.	Temperature test, -10°C operational soak	Test suspended; trouble disappeared when encoder analog oscillator card was moved. A thermocouple had been inserted between this card and adjacent card. The fit was apparently not correct when the cards were replaced.	Not considered to be a malfunction. Test repeated, however, with satisfactory results.
N-10	4/15/61	Ames Proton Analyzer	Unit fails to respond to excitation.	First pre-vacuum set-up operations	The unit had a broken internal lead. Twisting of the plug when connecting Ames Test Box broke the lead. During repair of broken lead, some insulating potting was accidentally removed and caused a shorted component. The component was again insulated.	
O-1	3/23/61	SUI GM Counter	Counter not functioning.	Prior to balance operations	GM tube replaced while payload on balance machine. Operation not checked.	
O-2	3/23/61	SUI GM Counter	Counter not functioning after test.	Spin test	None	

S-3 PROTOTYPE UNIT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
O-3	3/28/61	SUI GM Counter	Counter not functioning prior to test. Counter working after test.	Acceleration tests	Unit modified by insulating terminals with a Teflon strip. Possibility that a solder tip moved from the ground point causing corona discharge.	
O-14	5/1/61	SUI GM Counter	No readout.	Pre-test operations	Exact cause cannot be determined — two possibilities: (1) 700V lead to GM tube may have opened, (2) data out lead from tube may have shorted. Leads were lengthened by splicing and re-soldering. Leads then were potted at the terminals and joints to reduce possibility of corona.	
O-18	5/19/61	SUI GM Counter	At 80 microns, the current meter pegged and dropped the 6.89 regulated supply to 6.18 VDC. SUI meter turned off. Still malfunctioning at standard temperature and pressure.	Chamber evacuation	Test halted. One of the two switching transistors (2N 329A) in the primary of the converter failed (junction showed short). Flight spare counter installed in payload.	

APPENDIX H

TEST REPORT

PROTOTYPE UNIT VIBRATION TEST

1. SUMMARY

The S-3 Prototype Unit was subjected to vibration tests beginning March 29, 1961. The tests were conducted in accordance with the Environmental Test Plan, S-3 Energetic Particles Satellite dated February 14, 1961. * The objectives of the tests were:

a. To qualify the design of the electronic and structural components of the system under the rigors of vibration.

b. To obtain transmissibility data for future use in qualification or acceptance testing of individual subassemblies which may fail a system test or for acceptance testing of spare subassemblies.

During the first test (sinusoidal, thrust axis), the Transmitter and Ion-Electron Detector malfunctioned. A retest, after repairs, indicated satisfactory operation of these units. During subsequent vibration tests, the GM Telescope malfunctioned and several questionable transients occurred in the regulated voltage supplies to the experiments.

2. RESULTS

FIRST TEST: SINUSOIDAL VIBRATION, THRUST AXIS

The first resonance of the instrument platform occurred at 110 cps with accelerations of ±40 to ±100 g's at various locations.** The input acceleration at this frequency was ±11 g's. The Magnetometer Sensor experienced a Q=8:1 at 460 cps which decreased to 2.5:1 at 600 cps and 7:1 at 800 cps. The GM Telescope received ±55 g's (Q=1:1) at 2450 cps when other packages located nearby received 10% of that value. Other acceleration levels are noted in Figure H-2.

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* See Appendix D
** See Figures H-1 through H-14

NOTE: The transmissibility information presented in this appendix was not spectrum analyzed for harmonic components in the waveform.

During the test, the Transmitter current decreased from a normal 400 ma to 150 ma (at approximately 1000 cps) and remained there after the test was completed. The Transmitter output power had decreased to 150 milliwatts from its nominal 1.5 watts. It was later found that a weld joint in the final amplifier tube was defective. The tube was replaced prior to any further testing. (This Transmitter had been previously vibration tested as a subassembly with satisfactory results).

The Ion-Electron Detector's photomultiplier tube also malfunctioned during the first vibration test. The tube was replaced before the next test.

SECOND TEST: SINUSOIDAL VIBRATION, THRUST AXIS (RETEST)

No malfunctions occurred during this test and the transmitted acceleration levels were within 5% of those recorded in the previous test.

However, during this test at approximately 85 cps, the dc output voltages from the Regulator Converter experienced a transient of about 60 milliseconds duration. The input battery voltage did not fluctuate during this transient nor did the 26 vac line. After the test, all output voltages were within tolerances and no permanent degradation of the Regulator Converter was discovered.

REMAINING TESTS:

After completion of the full series of thrust-axis tests (random and combustion resonance, in addition to those already discussed) it was found that one of the geiger tubes of the GM Telescope was non-operative. Since no replacement tubes were available, the leads from the GM Telescope to the Encoder were switched from the defective tube to the operative tube. This allowed checkout and monitoring of at least part of the Telescope.

During portions of subsequent tests, the 6.5 v output line of the Regulator Converter was not within tolerances. Post-test checkout failed to reveal permanent degradation or reoccurrence of the intermittent condition.

The combustion resonance (600 cps) thrust axis test was conducted by subjecting the Prototype Unit to the maximum shaker (MB C-50) output within the 550 cps to 650 cps bandwidth. The input acceleration during this test was ±56 g at 550 cps and increased to ±62 g at 650 cps. No spacecraft failures were encountered.

The levels of acceleration experienced by the electronics during all the transverse-axes tests were generally less than Q=1.5:1

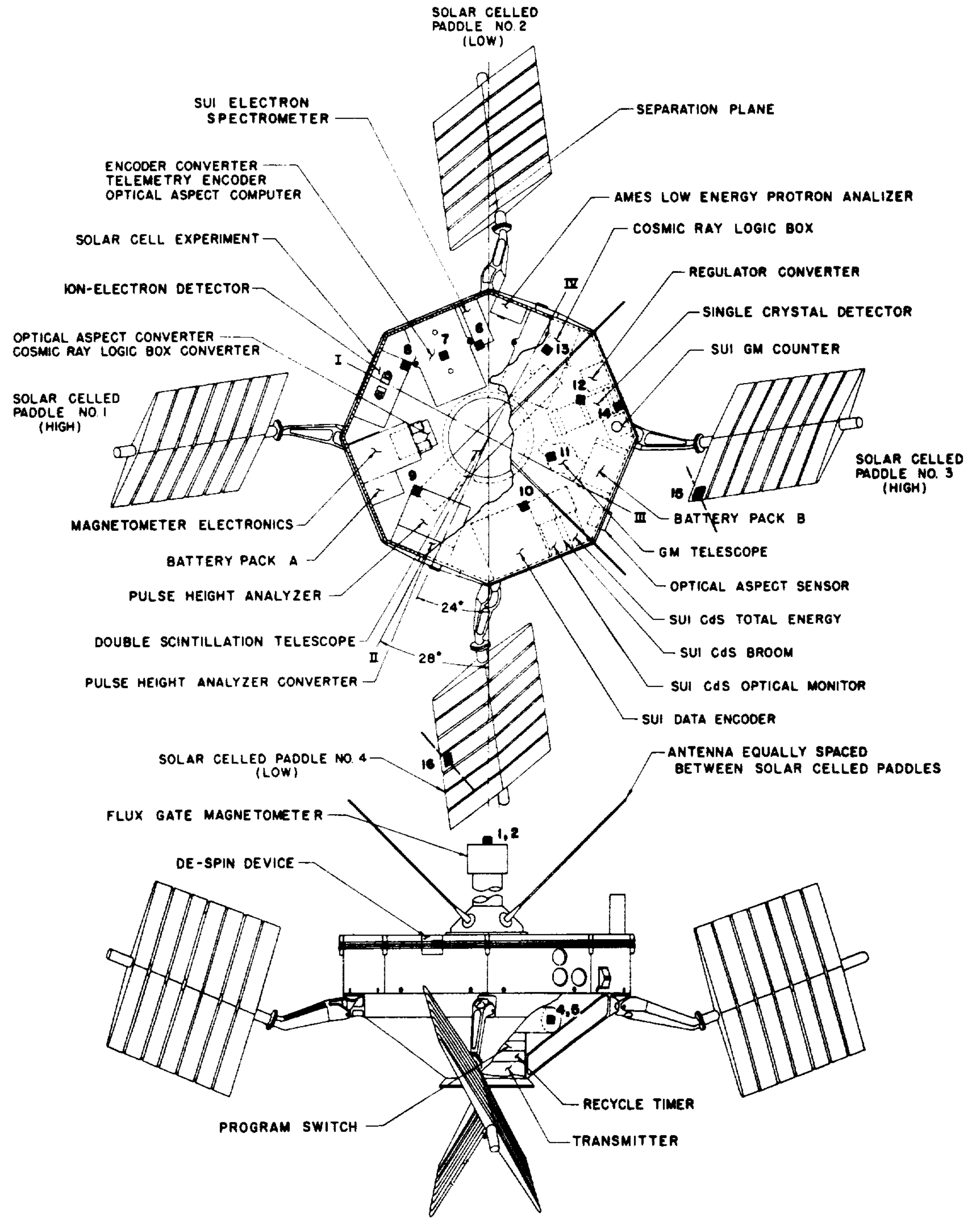
3. VIBRATION RETEST

On June 24, 1961, the S-3 Prototype Unit was subjected to vibration retests. The purpose of these tests was to qualify all those subassemblies which had been replaced, redesigned, repaired, or modified since the initial series of vibration tests.

Only thrust axis sinusoidal and random vibration tests were conducted.

The Prototype Unit passed the tests satisfactorily except for the GM Telescope which failed during the shaker equalization process. No data is available concerning the acceleration levels experienced by the Telescope during this process.

It was decided that Qualification testing of the GM Telescope would be continued separately.



NOTE: ACCELEROMETERS 6 THRU 13 WERE MOUNTED ON THE UNDER SIDE OF THE INSTRUMENT PLATFORM

NO. 14 WAS MOUNTED TO THE INNER SIDE OF THE PLATFORM BESIDE THE SUI GM COUNTER

NO'S 15, 16 WERE NORMAL TO THE PLANE OF THE PADDLES

NO'S 17, 18 & 19 WERE MOUNTED ON THE FIXTURE INTERFACE

NO 19 WAS THE INPUT ACCELEROMETER

ALL ACCELEROMETERS WERE ORIENTED IN THE THRUST DIRECTION EXCEPT NO. 2, 14, 15, 17 (I–III AXIS) AND NO. 3, 16, 18 (II–IV AXIS)

SEE FIGURE H-2 FOR RECORDED ACCELERATION LEVELS

Figure H-1. Location of Accelerometers
Prototype for Unit Vibration Tests

FREQ. (cps)	INPUT ACCELEROMETERS															
	#19	#1	#2	#4	#5	#6	#7	#8	#9	#10	#11	#12	#13	#14	#15	#16
55	±11g	±11g	±3g	±12g	±2g	±14g	±14g	±14g	±11g	±13g	±13g	±12g	±15g	±3g	±15g	±25g
105	±11g	±19g	±15g	±35g	±12g	±100g	±80g	±80g	±50g	±80g	±40g	±40g	±40g	±35g	±40g	±30g
176	±11g	±14g	±5g	±20g	±7g	±9g	±9g	±10g	±5g	±8g	±38g	±12g	±7g	±11g	±5g	±2g
300	±11g	±20g	±5g	±78g	±60g	±7g	±14g	±3g	±5g	±3g	±7g	±4g	±15g	±3g	±3g	±11g
460	±11g	±90g	±25g	±10g	±35g	±15g	±10g	±4g	±3g	±3g	±10g	±2g	±3g	±2g	±7g	±5g
600	±21g	±55g	±20g	±10g	±30g	±10g	±3g	±8g	±6g	±3g	±3g	±3g	±4g	±10g	±18g	±2g
800	±21g	±150g	±20g	±5g	±10g	±10g	±3g	±8g	±6g	±3g	±3g	±8g	±5g	±25g	±4g	±1g
2450	±54g	±5g	±2g	±8g	±4g	±8g	±10g	±5g	±10g	±10g	±55g	±20g	±2g	±3g	±2g	±1g

Figure H-2. Acceleration Levels Recorded During Sinusoidal (Thrust Axis) Vibration Test Prototype Unit

See Figure H-1 for locations
of individual accelerometers

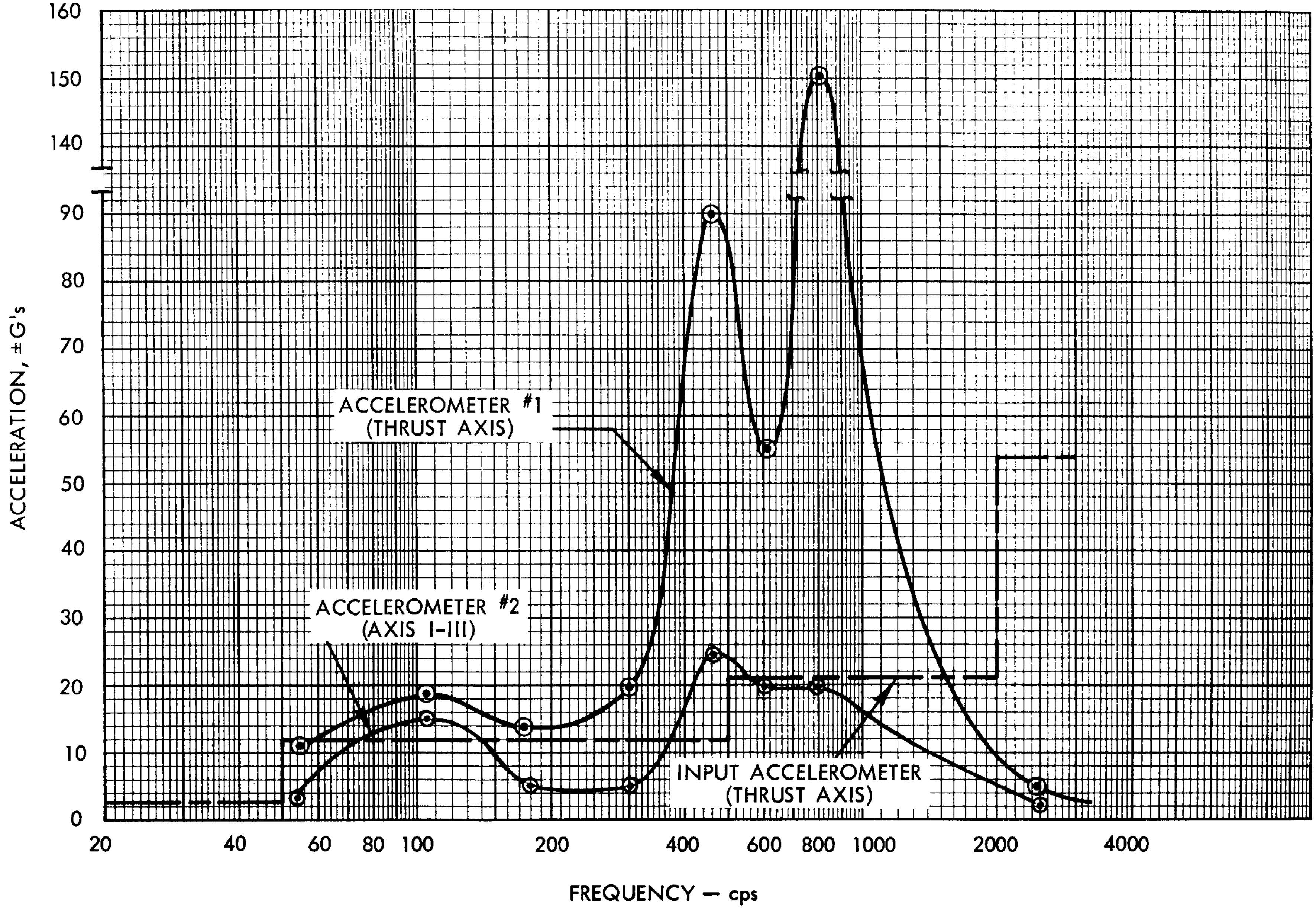


Figure H-3. Magnetometer Response

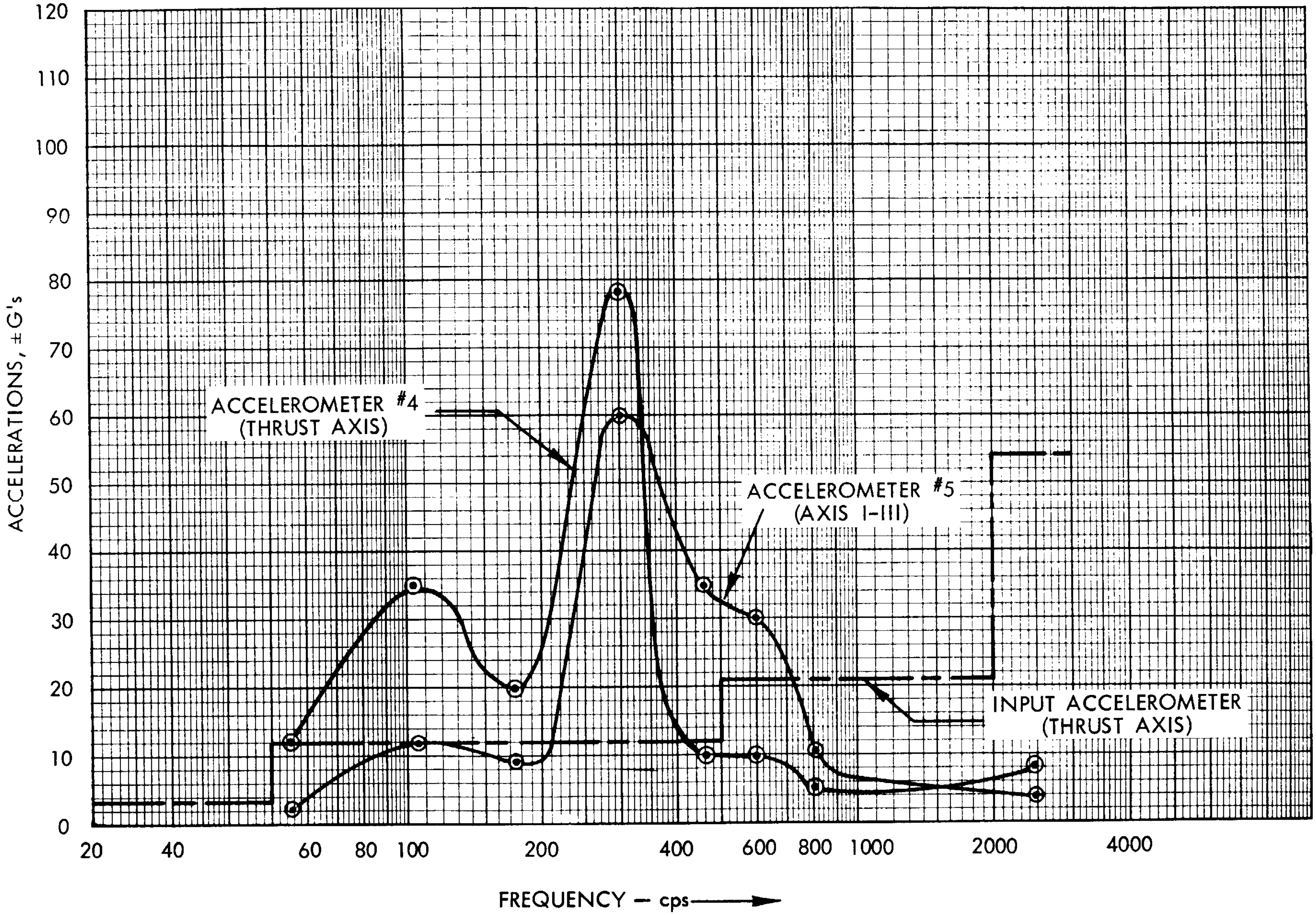


Figure H-4. Double Telescope Response

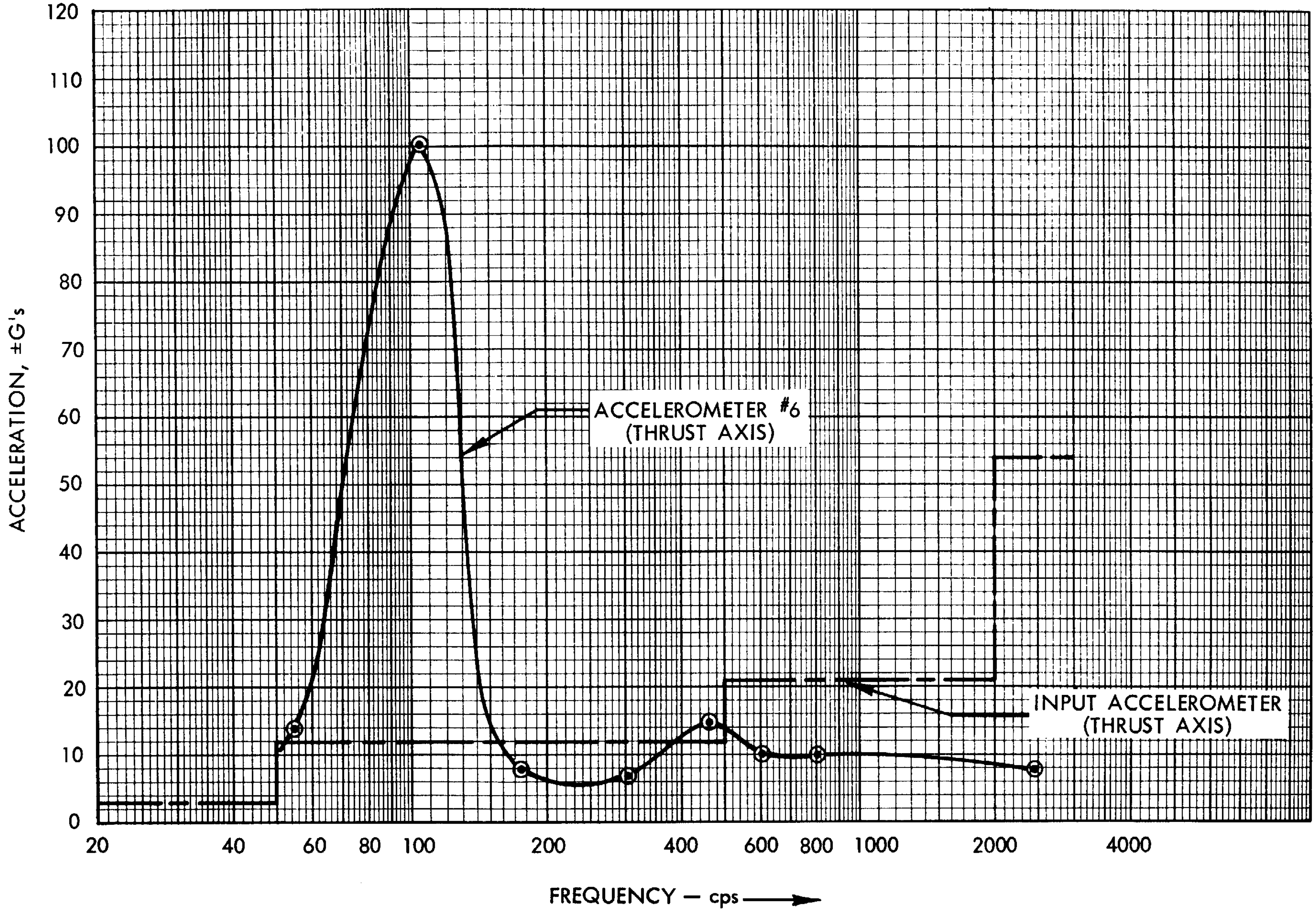


Figure H-5. SUI Electron Spectrometer Response

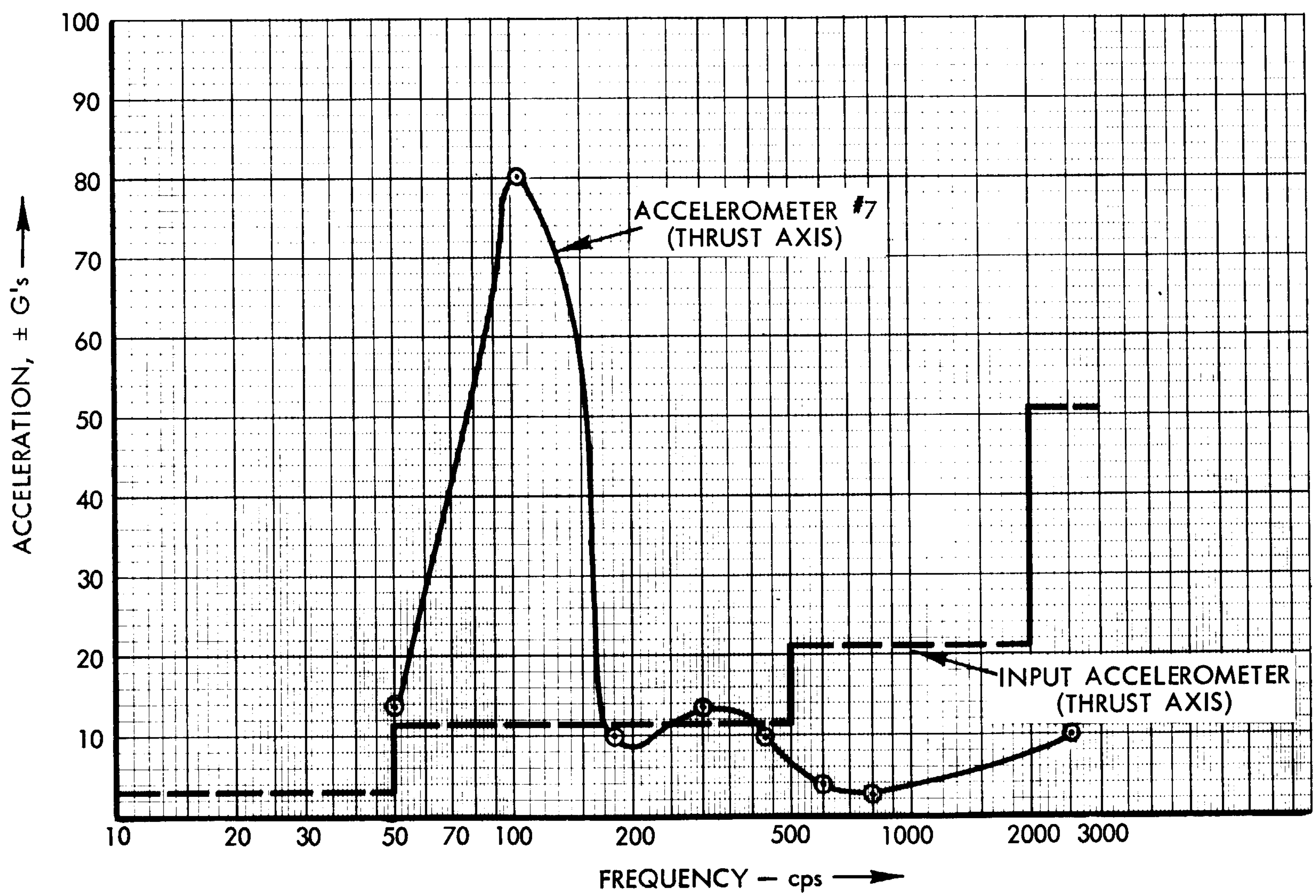


Figure H-6. Telemetry Encoder Response

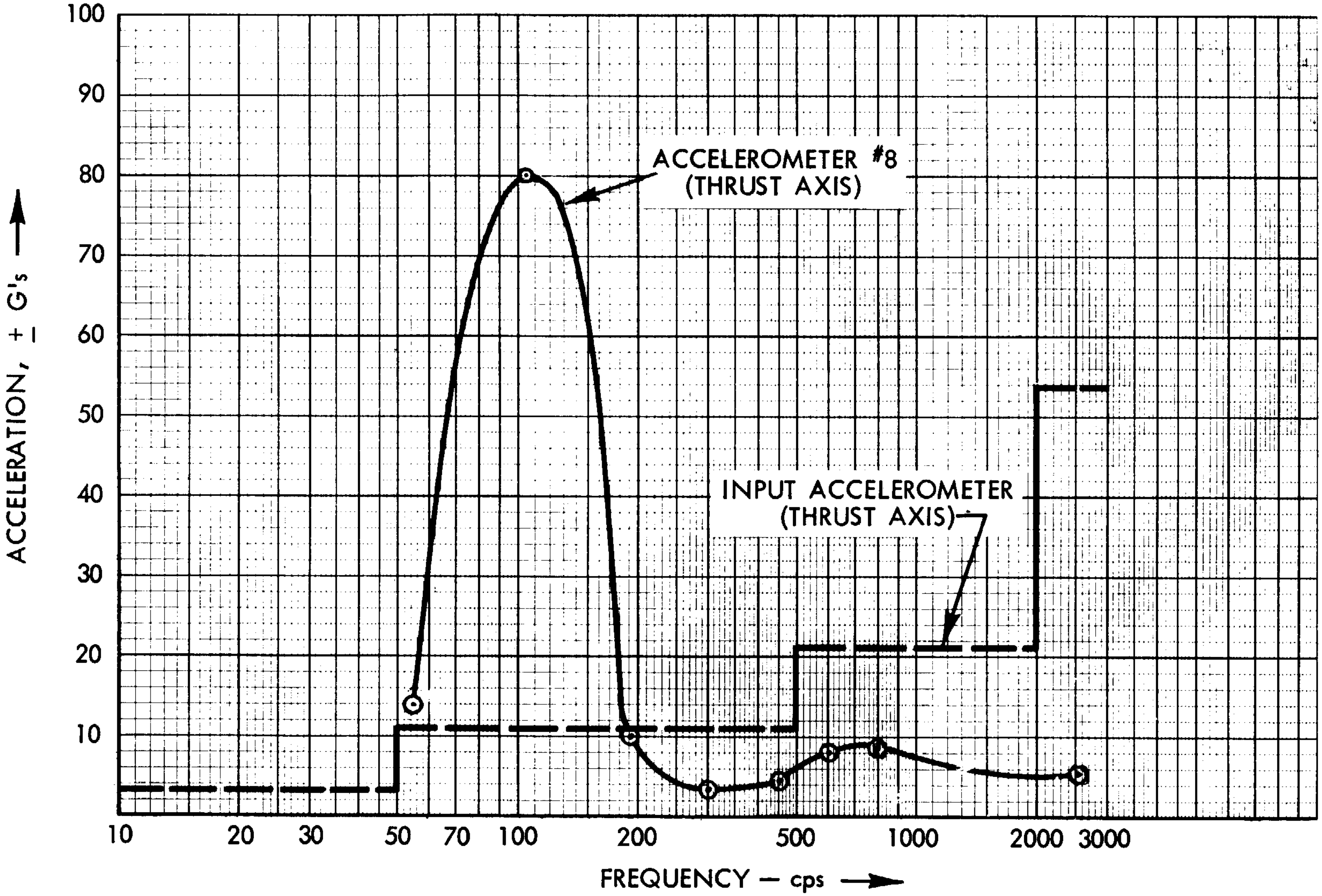


Figure H-7. Ion-Electron Detector Response

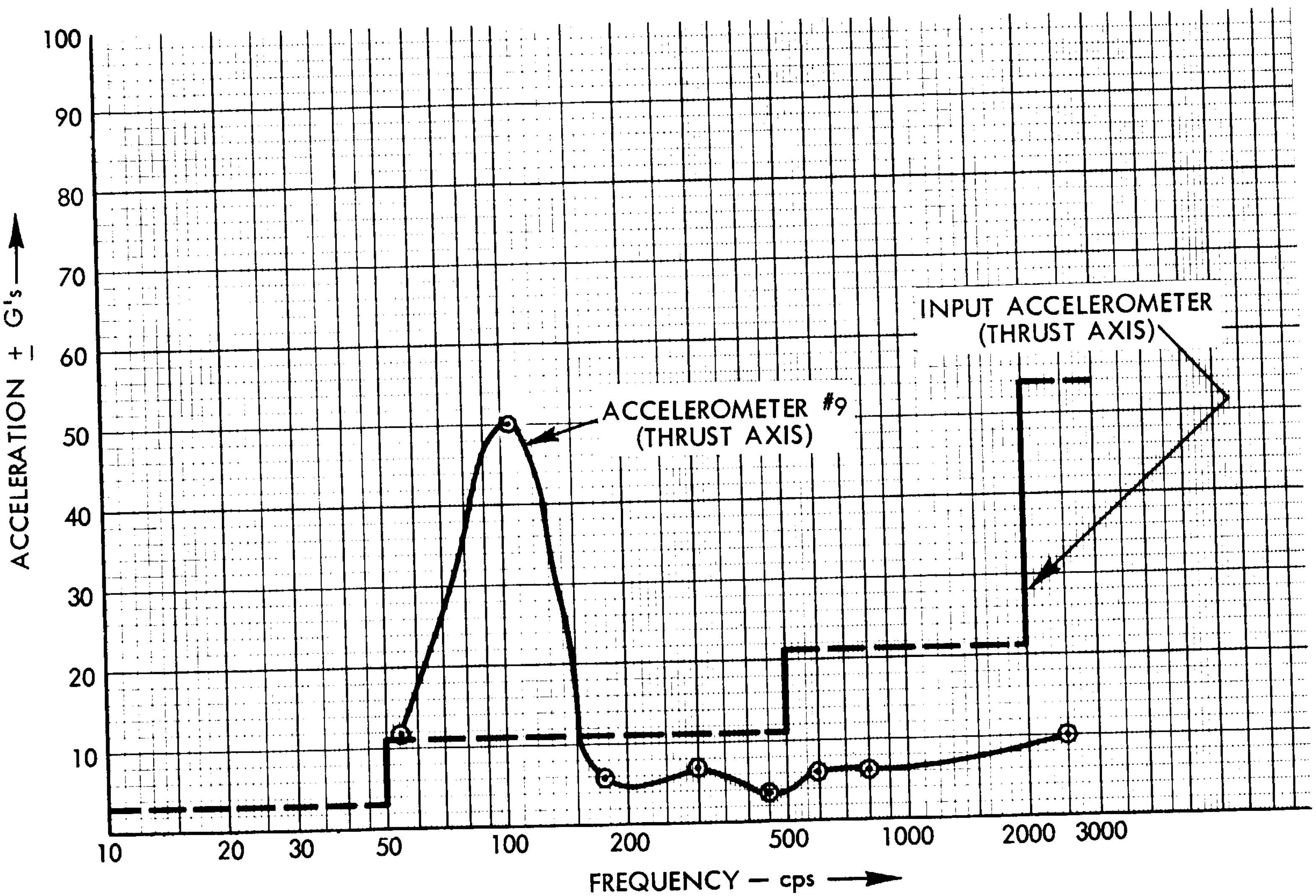


Figure H-8. Pulse-Height Analyzer Response

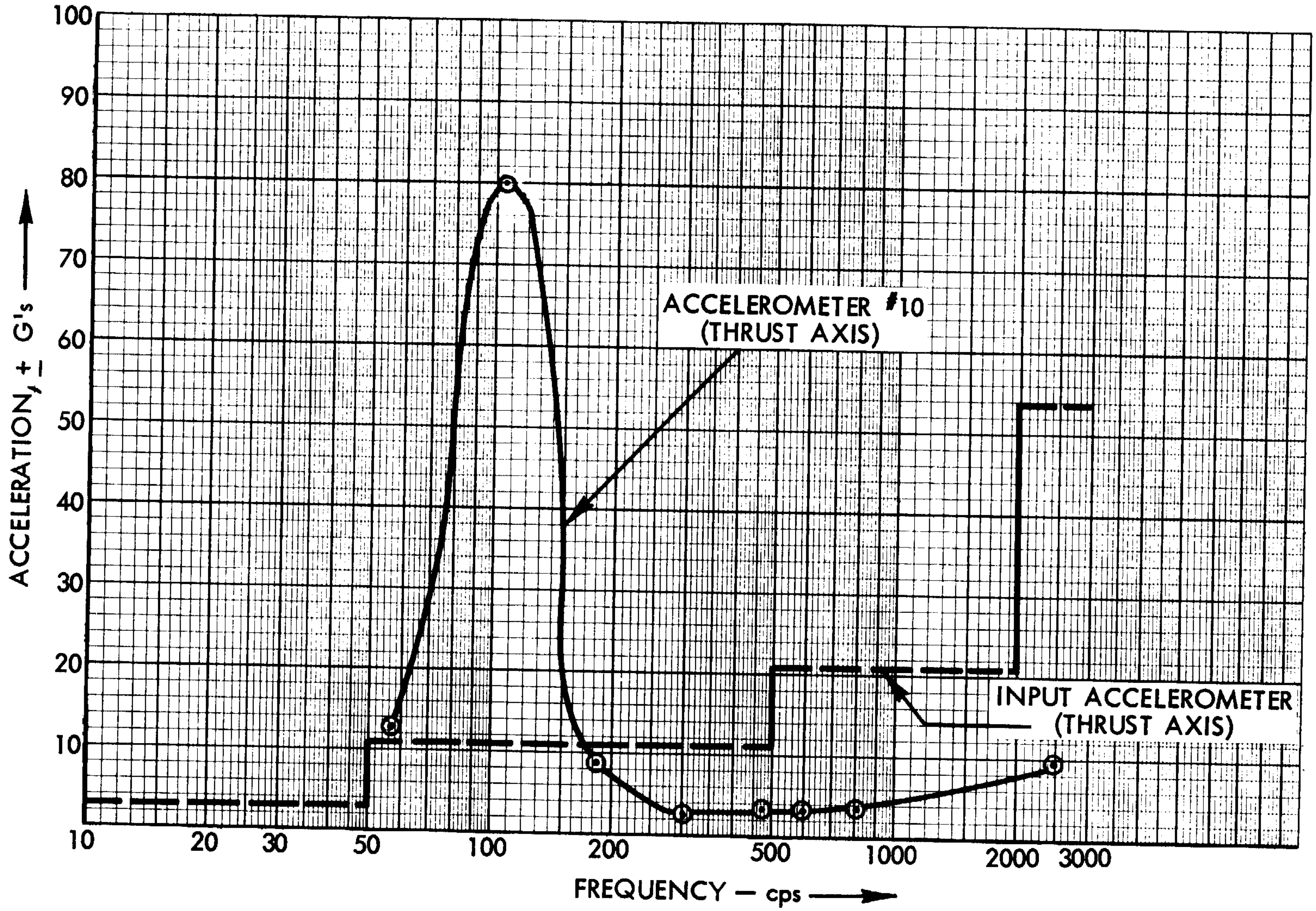


Figure H-9. SUI Data Encoder Response

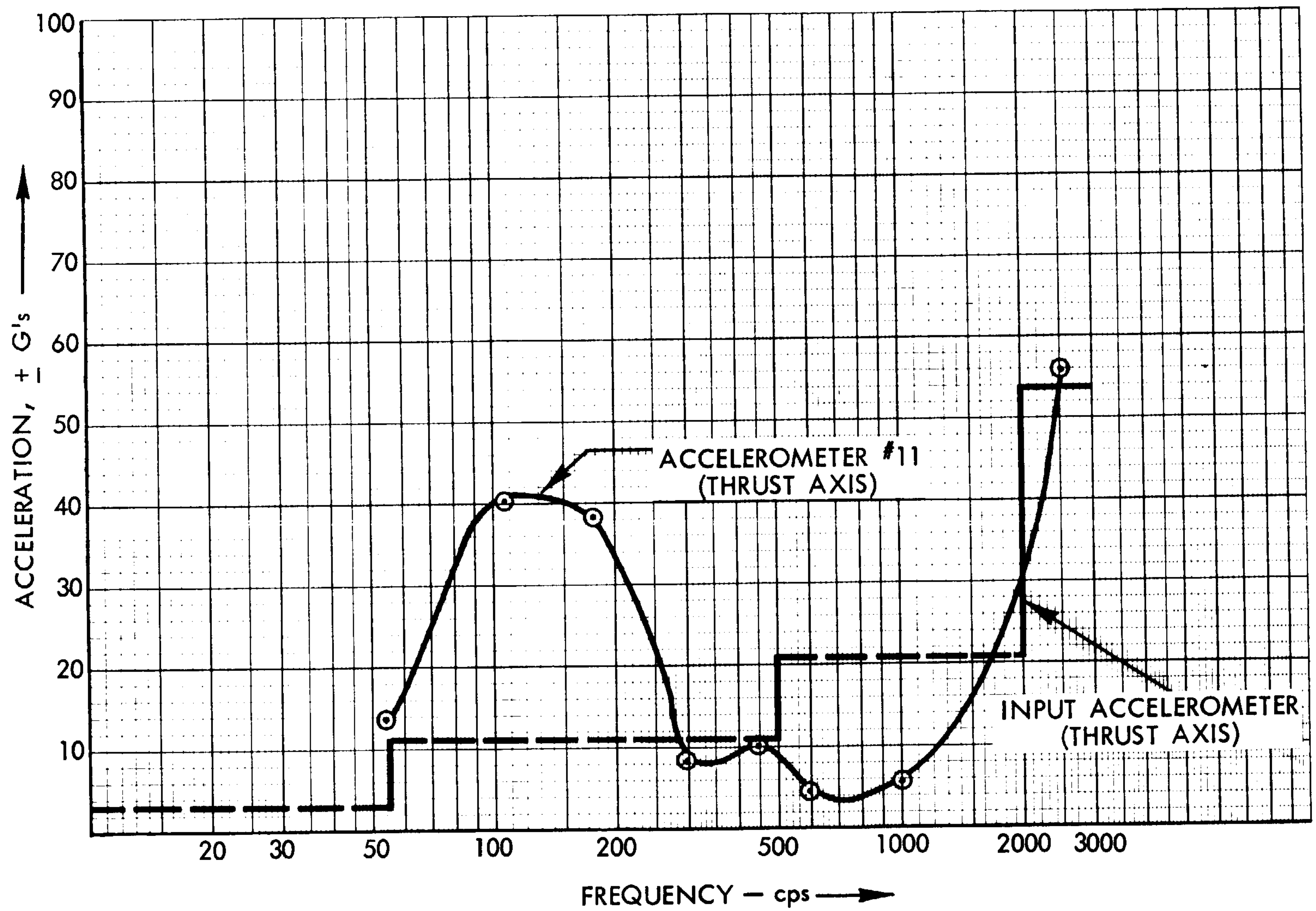


Figure H-10. GM Telescope Response

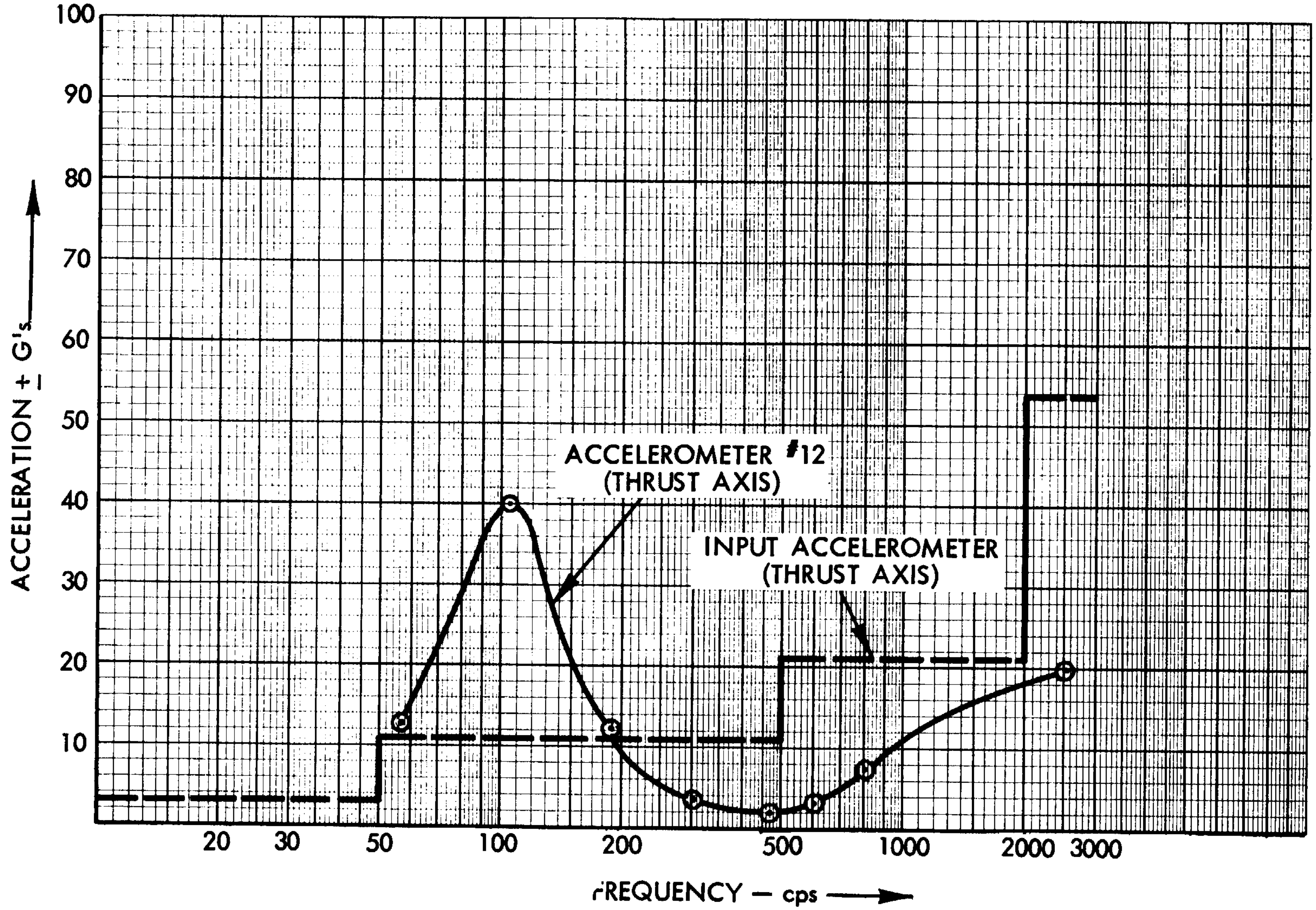


Figure H-11. Single Crystal Detector Response

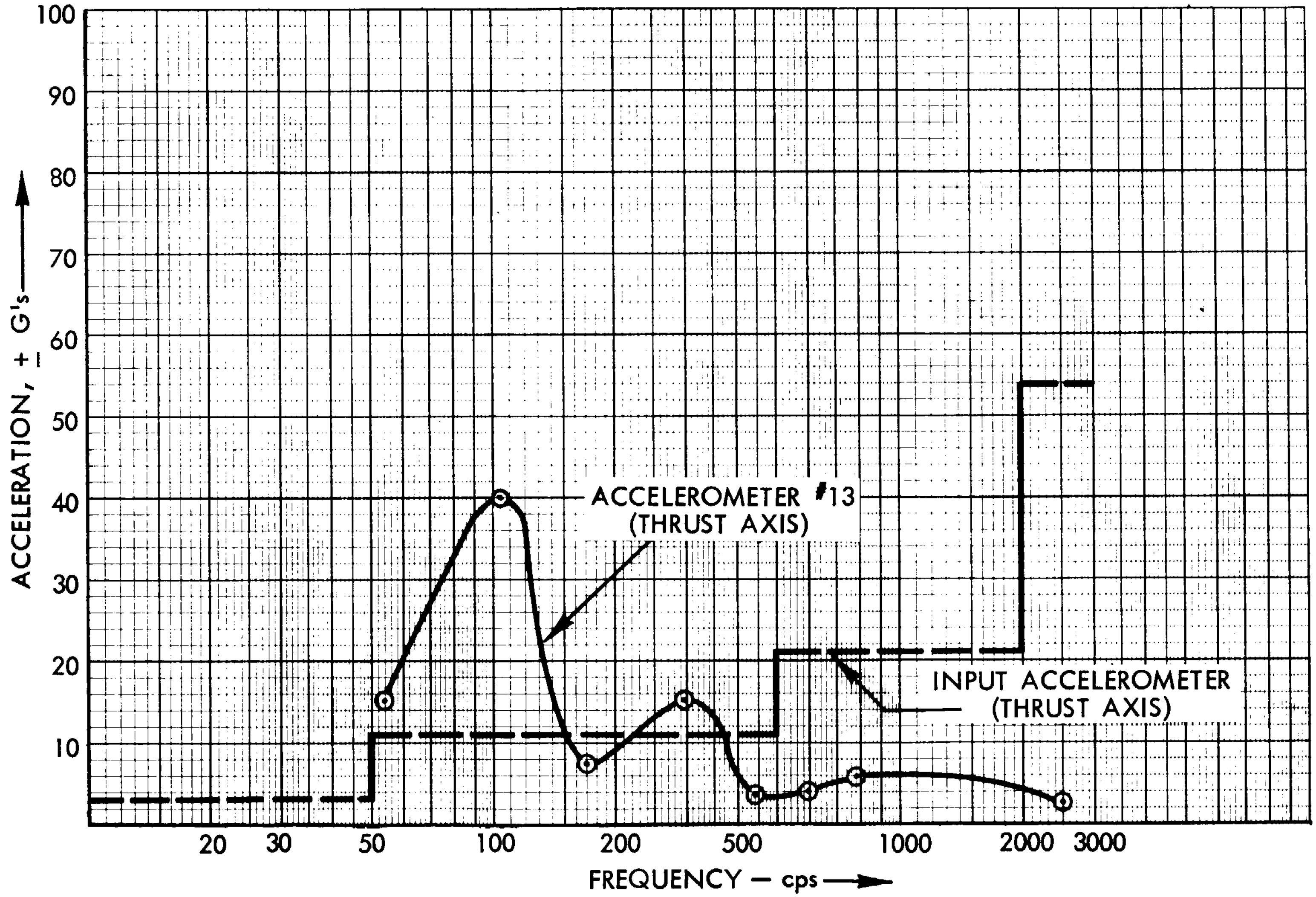


Figure H-12. Cosmic Ray Logic Box Response

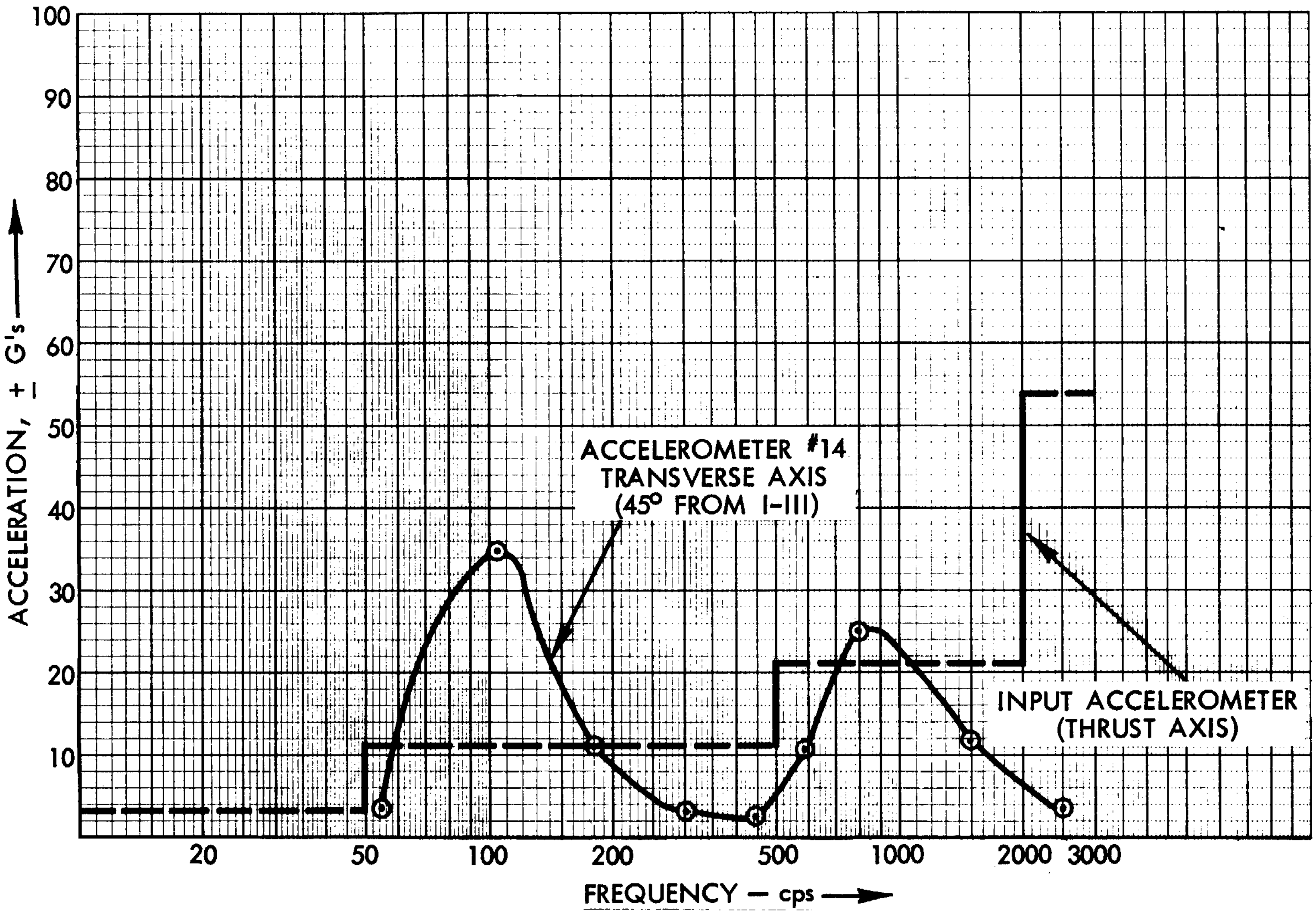


Figure H-13. SUI GM Counter Response

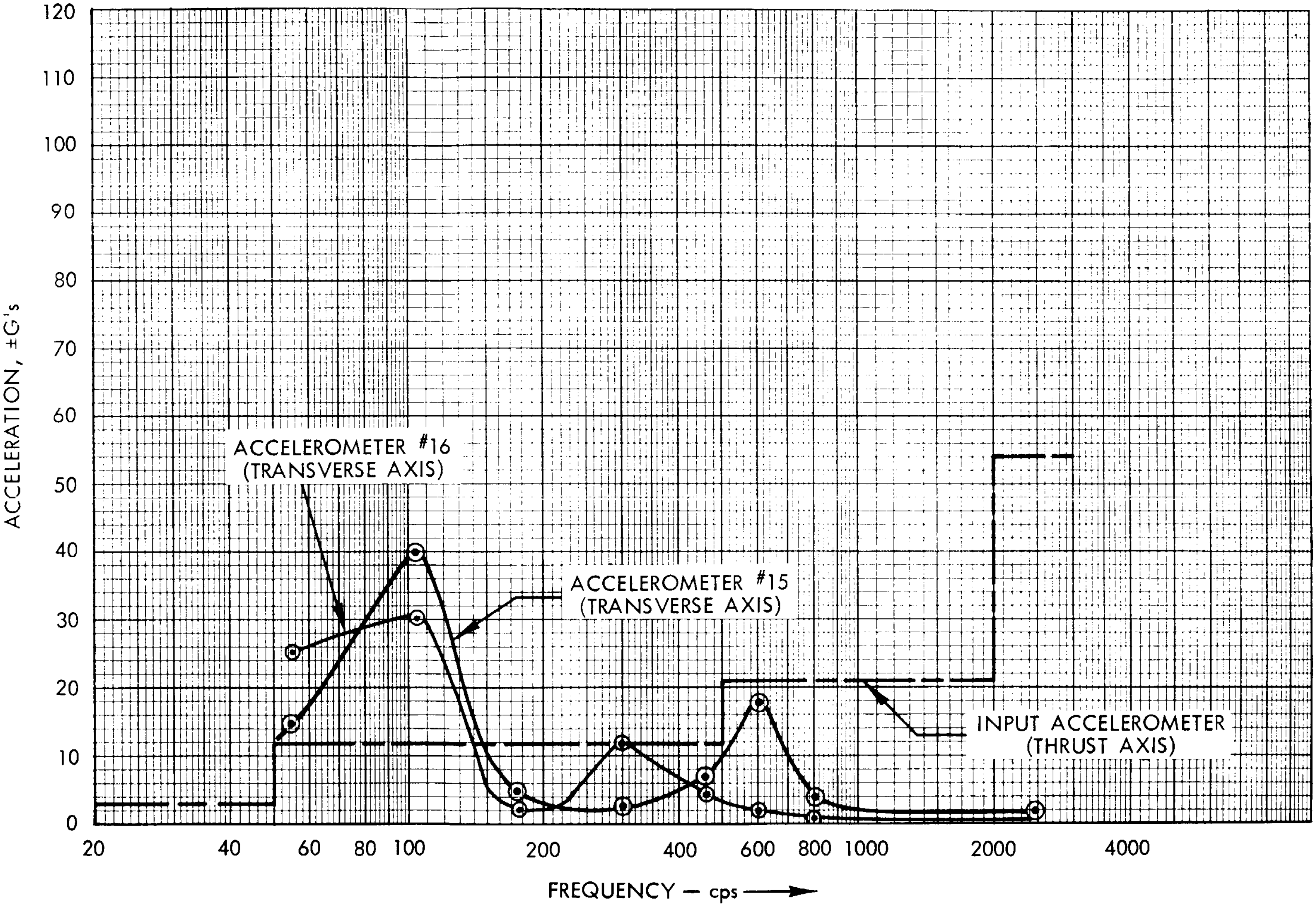


Figure H-14. Solar Paddle Response

APPENDIX J

TEST REPORT

PROTOTYPE UNIT TEMPERATURE TESTS

APPENDIX J: TEST REPORT PROTOTYPE UNIT TEMPERATURE TESTS

1. SUMMARY

The S-3 Prototype Unit was exposed to a series of temperature tests beginning April 4, 1961. The tests were conducted in accordance with the Environmental Test Plan, as modified.*

No malfunctions were observed during these tests although numerous operational difficulties were encountered.

2. TEST RESULTS

Pre-Temperature Checkout:

A systems checkout was performed at ambient temperature. It was noted at this time that when external power was applied to the spacecraft, the voltage dropped from 19v to approximately 4 volts. Further investigation revealed a short-circuit within the checkout-instrumentation leads. The short circuit was corrected by replacing a 37 pin connector.

Temperature Storage Tests:

The spacecraft (non-operational) was subjected to six hour tests at temperatures of -10°C and +50°C. Upon completion of the soak periods, the temperature of the spacecraft was lowered to +30°C for an operational check. Due to r.f. interference experienced using the spacecraft antennas within the chamber, it was decided to perform two checkouts; one using an external antenna and the second using the antennas inside the chamber.

a. External Antenna Checkout

The Prototype Unit was operated on external power. All voltages and currents were normal. It was observed, however, that when the relays controlling the compressors and heaters of the Temperature Chamber would close,

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* See Appendix D

the spacecraft would be turned off. It is believed this was due to a transient line drop caused by the circuitry of the chamber.

b. Internal Antenna Checkout

The spacecraft was operated on external power. During this checkout, a large amount of r.f. interference appeared to be causing the readout of Channel 4 (SUI) to be erratic. Also, the Proton Analyzer and Ion-Electron Detector experiments appeared to be affected by the r.f. being reflected within the stainless steel temperature chamber.

Low Temperature (Operational) Exposure:

The temperature of the chamber was lowered to -10°C. When thermal stabilization was attained, an attempt was made to operate the spacecraft on internal power, however, the Program Switch would not respond. The spacecraft was then turned on external power and the performance of the experiments checked. At this time, it was observed that improper data from the Ames Proton Analyzer was being received. It was then decided to increase the temperature of the chamber to ambient and attempt to locate the cause of the malfunctions.

When at ambient temperature, the chamber was opened and the meter panel connected directly into the turn-on plug. However, the spacecraft would not respond to internal power. Further investigation indicated that the Current Sensor had an open circuit between the Battery and the experiments. This was the result of the short circuit observed within the instrumentation leads during the test set-up. The faulty operation of the Ames Proton Analyzer was attributed to an intermittent connection between the experiment and the Telemetry Encoder. The intermittent connection was found to be due to the fact that the Telemetry Encoder had not been re-tightened securely after the installation of one thermocouple* within the stack. The faulty Current Sensor was replaced with the unit from the Flight Spacecraft.

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* See Figure J-1 - Location of Thermocouples

High Temperature (Operational) Exposure:

The spacecraft was replaced within the temperature chamber and a brief operational check indicated all experiments to be operating on both external and internal power. All currents and voltages were normal.

The temperature was then stabilized at +35°C and an operational check indicated that spacecraft was functioning normally.

Upon completion of the +35°C test, the temperature (while operating) was increased until the temperature of the warmest experiment (Regulator Converter) reached +50°C. When this occurred, a brief checkout indicated all experiments to be functioning properly. During this checkout the temperature of the Regulator Converter increased from +50°C to +56°C. *

Low Temperature (Operational) ReTest:

The temperature of the spacecraft was stabilized at -10°C to verify the operation of the Ames Proton Analyzer and also the capability of the spacecraft to be operated on internal power.

When at the stabilized temperature, it was found that the potential of the batteries was less than the minimum level required for operation (normally 12.8v). The Batteries employed will operate the payload approximately four hours depending on the amount of charge; approximately three hours had been logged on the batteries prior to this portion of the temperature test.

The Ames Proton Analyzer was operated on external power and performed satisfactorily.

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* See Table J-1 - Thermal Data

FIGURE J-1

THERMOCOUPLE LOCATIONS

No.	Location
1	Telemetry Encoder - between Analog and Digital Oscillators
2	Electron Spectrometer
3	Ames Proton Analyzer
4	Cosmic Ray Logic Box
5	Regulator Converter
6	Single Crystal Detector
7	SUI AM Counter
8	GM Telescope
9	Battery Pack "A"
10	Optical Aspect Sensor
11	CDS Total Energy (SUI)
12	Data Encoder (SUI)
13	Current Sensor
14	Pulse Height Analyzer Converter
15	Pulse Height Analyzer
16	Battery Pack "B"
17	Magnetometer Electronics
18	Optical Aspect Convert and Cosmic Ray Logic Converter
19	Ion-Electron Detector
20	Double Telescope
21	Transmitter (outside)
22	Transmitter - between Transmitter and Recycle Timer
24	Chamber Ambient

TABLE J-1

THERMAL DATA

1. Low Temperature Soak (-10°C, 6 hours)

Time Started: 0010 4-5-61
Time Completed: 0610 4-5-61

Thermocouple Location:	Temperatures At Start	At End
Telemetry Encoder	26.5°C	-10°C
Transmitter	16.0°C	-10°C
Chamber Ambient	-11.0°C	-10°C

2. High Temperature Soak (+50°C, 6 hours)

Time Started: 0710 4-5-61
Time Completed: 1310 4-5-61

Thermocouple Location:	Temperatures At Start	At End
Telemetry Encoder	-1.0°C	50°C
Transmitter	26°C	51°C
Chamber Ambient	+50°C	50°C

3. Post-Soak Operation (+30°C)

Time Started: 1435 4-5-61
Time Completed: 1737 4-5-61

Thermocouple Location:	Temperatures At Start	At End
Telemetry Encoder	41.0°C	30.5°C
Regulator Converter	31.0°C	36.5°C
Transmitter	28.5°C	33.0°C
Chamber Ambient	28.5°C	28.5°C

4. <u>Low Temperature Operation</u> (-10°C)

Time Started: 1020 4-6-61
Time Completed: 1145 4-6-61

Thermocouple Location:	Temperatures At Start	Temperatures At End
Telemetry Encoder	-10.5	-10.0
Regulator Converter	-10.5	-5.0
Transmitter	-10.5	-7.5
Chamber Ambient	-10.5	-10.5

5. <u>High Temperature Operation</u> (+35°C)

Time Started: 2140 4-6-61
Time Completed: 2400 4-6-61

Thermocouple Location:	Temperatures At Start	Temperatures At End
Telemetry Encoder	36.0°C	36.0°C
Regulator Converter	36.0°C	43.0°C
Transmitter	37.0°C	39.0°C
Chamber Ambient	35.0°C	35.0°C

6. <u>High Temperature Operation</u> (+50°C)

Time Started: 0100 4-7-61
Time Completed: 0200 4-7-61

Thermocouple Location:	Temperatures At Start	Temperatures At End
Telemetry Encoder	39.0°C	44.5°C
Regulator Converter	*50.0°C	56.0°C
Transmitter	49.0°C	52.5°C
Chamber Ambient	49.5°C	51.0°C

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* Used as control thermocouple in this test only.

APPENDIX K

TEST REPORT

PROTOTYPE UNIT THERMAL-VACUUM TEST

APPENDIX K: TEST REPORT PROTOTYPE UNIT - THERMAL VACUUM TEST

1. SUMMARY

The Prototype Unit was exposed to a series of thermal-vacuum exposures beginning April 14, 1961.

The test consisted of four exposures conducted in the following sequence:

a. High Temperature Soak
b. Low Temperature Soak
c. 45° Solar Aspect
d. 135° Solar Aspect

Exposure "c" simulated the space environment if the sun was at an angle of 45° to the longitudinal (spin) axis of the spacecraft and illuminating the top cover of the instrument compartment.

Exposure "d" simulated the space environment if the sun was at an angle of 135° to the longitudinal axis of the spacecraft and illuminating the bottom cover of the instrument compartment.

Exposures "a" and "b" were conventional tests during which the spacecraft was "soaked" at a uniform chamber temperature.

The test was conducted in accordance with the modifications* to the Environmental Test Plan - Energetic Particles Satellite (S-3).

Many problems developed during the test including the failure of the Solar Array Voltage Regulator, the over-heating of the Regulator Converter, and the inability of the Transmitter Converter to start at low temperatures. In addition, malfunctions of the following experiments were encountered: GM Telescope, Single Crystal Detector, Double Telescope Pulse Height Analyzer (2 occasions), Magnetometer Electronics and SUI GM Counter (2 occasions).

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* See Appendix D.

2. TEST CONDITIONS

Test Setup

Figure K-1 indicates the mounting configuration of the spacecraft during the thermal vacuum test. The light rings were utilized to establish the thermal gradient conditions for the 45° and 135° solar aspect exposures.

During the test, one "live" Solar Paddle was attached (at position No. 1) to the spacecraft, and two lamps were positioned to permit illumination of either or both sides of the paddle. Another lamp was positioned to illuminate the Solar Cell Damage Experiment. A single lamp was positioned to control the heating of the Transmitter compartment and still another to control the temperature of the Magnetometer sensor.

Figure K-2 shows the spacecraft and lamps installed in the GSFC 8' x 8' thermal-vacuum chamber.

Test Parameters

The high temperature (48 hour) and low temperature (48 hour) Vacuum-Soaks were performed at uniform chamber temperatures of +35°C and -10°C respectively.

During the 45° Solar Aspect exposure, the temperature of the top cover (station 7, Figure 1) was stabilized at +45°C with the Transmitter (stations 14, 21) cooled to -20°C. After the above conditions were established, the spacecraft was operated for 48 hours.

During the 135° Solar Aspect exposure, the temperature of the back cover (station 12) was established at +35°C, with the magnetometer tube (station 4) cooled to -10°C. With these conditions established, the spacecraft was operated for 48 hours. Throughout each solar aspect exposure, the temperature of the chamber wall was maintained at -55°C.

During the thermal-vacuum test the following minimum chamber pressures were attained:

a. +35°C Vacuum soak ...... 3.3×10^{-6} mmHg
b. -10°C Vacuum soak....... 2.1×10^{-7} mmHg
c. 45° Solar aspect....... 4.8×10^{-6} mmHg
d. +35° Solar aspect....... 5.3×10^{-7} mmHg

3. TEST RESULTS

Thirteen malfunctions and/or marginal operation of subassemblies were observed during the test - five in the spacecraft power and power program systems, and eight within individual experiments.

The following is a discussion of each of the discrepancies, the prevailing test conditions, and the action taken to resolve the problem. Appendix G summarizes these and other problems encountered during the Design Qualification testing of the Prototype Unit.

POWER AND POWER PROGRAM SYSTEMS

1. Solar Array Voltage Regulator

This device limits the voltage of the Solar Paddles at 19.6v. It also dissipates (as heat) excess power from the Paddles.

The failure occurred during the high temperature (+35°C) exposure and apparently was the result of thermal runaway of the power transistor. Associated with the thermal runaway of the transistor was a decrease of the impedance of the device which permitted discharge of the Batteries through the Current Sensor in the reverse direction. This in turn, caused the destruction of one of the windings of the Current Sensor and resulted in the isolation of the spacecraft from the Solar Paddles.

The device was redesigned to include a parallel combination of two power transistors each in series with a resistor. The successful operation of the redesigned Solar Array Voltage Regulator was verified during the subsequent high temperature vacuum soak exposure.

2. Regulator Converter

This device, the main converter in the spacecraft power system, supplies four regulated outputs to the spacecraft electronics: +6.5v, +12.0v, -17.8v and +26.2v AC (@2 KC). Input to the Converter is 13 to 12v from the Batteries or Solar Array.

When the spacecraft is operated using an external power supply, the supply is adjusted to 15.0v unless otherwise indicated.

During the high temperature exposure, the temperature of the Converter (as monitored by a thermocouple attached to its exterior) rose to 59.5°C and was still increasing when it was decided to increase the input voltage from 15.0v to 20v. The higher voltage resulted in higher efficiency (less power dissipation) and thereby decreased the operating temperature to +57°C.

When the Prototype Unit was returned to atmospheric conditions, a heat sink was added in the vicinity of the Regulator Converter and each electronic package within the instrument compartment was painted dull black.

The subsequent high temperature exposure indicated an operating temperature of +52°C with an input voltage of 15.0v--a decrease of at least 8°C.

With the application of the flat black paint to the electronics, the spacecraft operating temperatures appear to have been reduced approximately 2.0°C. The average temperature of the experiments, after 20 hours operation, was 42°C without the paint and 40°C with the paint. However, the temperature change observed may have been partially due to the addition of an aluminum baffle in front of the liquid nitrogen baffle of the chamber diffusion pump and a slight difference in chamber wall temperature.

3. Transmitter (Transmitter Converter)

During the low temperature exposure, the final stage of the Transmitter would not operate (temperature: -12.5°C). When the temperature was raised to -7°C, normal operation was observed.

This malfunction was not due to the Transmitter, but rather to the Transmitter Converter (located in the Transmitter card) which supplies +180v to the plate and +6.3v to the filament of the final amplifier--a Sylvania 5977 vacuum tube.

In order to correct this deficiency, the value of a "starting resistor" in the Transmitter Converter was changed.

The test was repeated, but once again (temperature -10°C) the final stage would not operate properly. The test was suspended, the Transmitter removed from the spacecraft and a bench check performed. This check did not indicate any deficiencies in the Transmitter or its converter. A subassembly test of the unit at -20°C, -10°C and 0°C in a vacuum indicated normal starting characteristics. A twelve hour soak at -20°C and 1×10^{-5} mmHg indicated no deficiencies.

It was surmised that the ground station monitoring equipment may have been responsible for the latter starting difficulties. The Transmitter was replaced in the Prototype Unit and the system testing resumed.

4. Program Switch

During the second high temperature soak, the under voltage lock out circuit failed to turn the spacecraft off. However, this condition occurred under unrealistic circumstances--operation on external power (13.0v), Batteries near complete discharge. When the power supply was interrupted momentarily, the spacecraft load was transferred to the Batteries (with the battery voltage just above the lock out point). The resulting rapid decrease of the battery voltage failed to turn the spacecraft off.

The Program Switch was modified to include additional amplification to the "off bus" and condenser storage added to hold-up the under-voltage circuit when the Batteries collapse.

In addition to the under-voltage problem, the Program Switch failed on several occasions to operate on command from the Blockhouse Control. Repeated attempts to command the Program Switch were successful, thereby indicating a marginal condition either in the switch itself or in the external command circuitry. This problem was not resolved.

5. Recycle Timer

During the high temperature exposure, one of the two oscillators in the Recycle Timer failed to activate the Program Switch. The redundant oscillator did operate the spacecraft.

This marginal turn-on characteristic was alleviated by the addition of a "pulse stretcher" in the timer circuitry.

INDIVIDUAL EXPERIMENTS

1. Single Crystal Detector

During the first high temperature soak, extensive arcing in the High Voltage Power supply (HVPS) produced interference with the Telemetry Encoder. When the HVPS failed completely some time later, the interference stopped. The detector was replaced with a spare unit.

2. Pulse Height Analyzer

A cold solder joint caused loss of data during the first high temperature soak.

3. Magnetometer Electronics

An excessive amount of current was drawn during the major portion of the high temperature soak test. Operation was normal during the subsequent low temperature soak test.

The failure was apparently caused by thermal runaway of two transistor amplifiers. These transistors were replaced and the bias resistors changed to decrease the possibility of thermal runaway.

4. SUI GM Counter

No data could be obtained from the GM Counter during a pre-test checkout (at atmospheric conditions).

The exact cause could not be determined but was suspected to be due to either an open circuit from the 700v line to the tube or a short circuit of the "data-out" lead. These circuits were rewired.

5. Pulse Height Analyzer

A defective transistor in the decoding gate caused a loss of readout during the low temperature soak test. The transistor was replaced.

6. GM Telescope

During a system checkout prior to the start of the solar aspect exposures, the second tube of the telescope was determined to be non-operative. (The first tube was non-operative since the vibration test).

7. Double Telescope

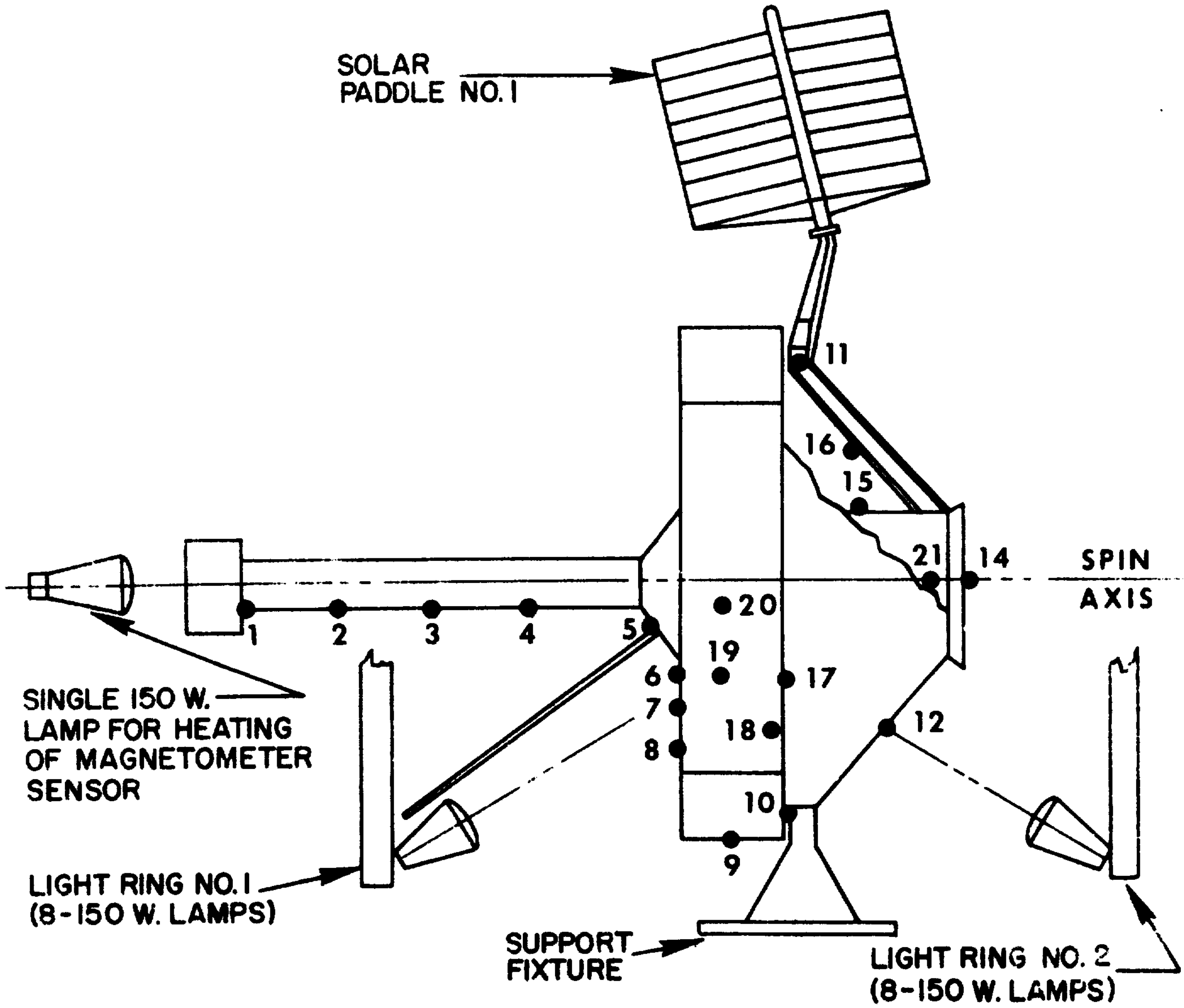
The power supply of the Telescope malfunctioned during the 45° Solar Aspect exposure. The power supply was replaced after the completion of the thermal vacuum tests.

8. SUI GM Counter

During chamber evacuation (pressure, 80 microns), one of the switching transistors in the primary of the (SUI) converter shorted. This short reduced the regulated supply from the Regulator Converter from 6.9v to 6.2v. The entire counter was replaced with a spare unit.

4. PROTOTYPE UNIT VACUUM RETEST

A retest of the Prototype Unit began June 25, 1961. The 48 hour vacuum soak at ambient temperature indicated acceptable operation of all subassemblies.



NOTE: Numbers refer to stations for which temperatures have been calculated by the Thermal Systems Branch. Skin temperature locations are indicated as dots attached to a line.

Figure K-1. Setup for Thermal-Vacuum Tests of the Prototype Unit

TABLE K-1

Summary of Thermal Data
Prototype Unit

(degrees Centigrade)

POSITION	45° SOLAR ASPECT			+35°C VACUUM SOAK			-10°C VACUUM SOAK			135° SOLAR ASPECT		
	B	A	ΔT	B	A	ΔT	B	A	ΔT	B	A	ΔT
1. Telemetry Encoder	0	33.5	33.5	35.5	38.0	2.5	-10.0	-2.0	8.0	-1.0	7.0	8.0
2. SUI Electron Spectrometer	5.0	33.0	28.0	35.5	37.0	1.5	-10.0	-2.5	7.5	-3.0	5.0	8.0
3. Ames Proton Analyzer	14.0	38.0	24.0	35.0	37.0	2.0	-10.0	-2.5	7.5	-7.0	1.5	8.5
4. Cosmic Ray Logic Box	18.5	42.0	23.5	35.0	37.5	2.5	-10.0	-1.0	9.0	-5.0	4.5	9.5
5. Regulator Converter	4.0	44.0	40.0	35.0	59.0	24.0	-10.0	13.0	23.0	-5.0	18.0	23.0
6. Single Crystal Detector	7.0	37.0	30.0	35.0	41.0	6.0	-10.0	2.0	12.0	-1.5	10.0	11.5
7. SUI GM Counter	10.5	35.0	24.5	35.0	38.0	3.0	-10.0	-2.0	8.0	-5.5	1.5	7.0
8. Battery Pack 'B'	8.5	36.0	27.5	35.0	38.0	3.0	-10.0	-2.5	7.5	-0.5	6.5	7.0
9. GM Telescope	12.0	37.5	25.0	35.0	38.0	3.0	-10.0	-2.5	7.5	-0.5	7.5	8.0
10. Optical Aspect Sensor	7.0	29.0	22.0	35.5	37.0	1.5	-10.0	-6.0	4.0	-6.0	-1.5	4.5
11. SUI Cadium Sulfide	13.0	37.0	24.0	35.5	37.0	1.5	-10.0	-3.5	6.5	-1.5	6.0	7.5
12. SUI Data Encoder	26.0	47.0	21.0	35.5	37.0	1.5	-10.0	-3.5	6.5	-5.0	2.5	7.5
13. Pulse Height Analyzer Converter	21.0	43.5	22.5	35.5	37.5	2.0	-10.0	-2.5	7.5	-8.0	0	8.0
14. Pulse Height Analyzer	23.0	45.0	22.0	35.5	37.5	2.0	-10.0	-2.5	7.5	-7.5	0	7.5
15. Battery Pack 'A'	4.0	30.0	26.0	35.5	37.5	2.0	-10.0	-3.5	6.5	0	7.0	7.0
16. Mangetometer Electronics	24.0	43.5	19.5	35.5	37.0	1.5	-10.0	-3.5	6.5	-7.5	-2.0	5.5
17. Optical Aspect Converter Cosmic Ray Logic Box Converter	8.0	33.0	25.0	35.5	37.5	2.0	-10.0	-2.5	7.5	2.0	10.0	8.0
18. Ion Electron Detector	13.0	37.0	24.0	35.5	37.0	1.5	-10.0	-4.5	5.5	-8.0	-1.5	6.5
19. Between Recycle Timer and Transmitter	-15.0	21.5	36.5	35.5	45.0	9.5	-10.0	11.0	21.0	27.5	50.5	23.0
20. Transmitter (outside cover plate)	-15.0	17.5	32.5	35.5	42.0	6.5	-10.0	6.5	16.5	32.5	50.5	18.0
21. Magnetometer Sensor	25.0	35.5	10.5	35.5	37.0	1.5	-13.0	-10.0	3.0	-11.0	-13.0	-2.0
22. Station 7	45.0	56.5	11.5	34.0	35.0	1.0	-12.0	-4.0	8.0	-20.5	-8.0	12.5

Key:
B = Before spacecraft operation
A = After spacecraft operation, temperature stabilized
ΔT = (A-B) Temperature rise due to spacecraft operation

APPENDIX L

PROBLEM AREAS ENCOUNTERED

DURING

ENVIRONMENTAL TESTING

OF THE

S-3 FLIGHT SPACECRAFT AND FLIGHT SPARE SPACECRAFT

PROBLEM AREAS ENCOUNTERED
DURING THE
ENVIRONMENTAL TESTING OF THE
S-3 FLIGHT SPACECRAFT & FLIGHT SPARE SPACECRAFT

JULY 21, 1961

		ITEM	1 BALANCE	2 PRE-VIBRATION	3 VIBRATION	4 FIRST PRE-TEST SET-UP AND CHECKOUT	5 CHAMBER EVACUATION	6 30° SOLAR ASPECT (FLIGHT SPARE ONLY)	7 SECOND PRE-TEST SET-UP AND CHECKOUT	8 30° SOLAR ASPECT	9 THIRD PRE-TEST SET-UP AND CHECKOUT	10 30° SOLAR ASPECT	11 150° SOLAR ASPECT	12 COLD SOAK (-10°)	13 COLD SOAK RETEST (FLIGHT SPARE ONLY)	14 SUB-ASSEMBLY ACCEPTANCE TESTING REQUIRED
			22 MAY	27 MAY	27 MAY	29 MAY	1 JUNE		3 JUNE	4 JUNE	4 JUNE	5 JUNE	8 JUNE	10 JUNE		
FLIGHT UNIT	A	SYSTEM CONVERTER		$\boxed{C}$												NO
FLIGHT UNIT	B	rf CHOKE CIRCUIT				☒ $\boxed{R}$										NO
FLIGHT UNIT	C	DOUBLE TELESCOPE				◪ $\boxed{C}$										NO
FLIGHT UNIT	D	GM COSMIC RAY TELESCOPE					◪ $\boxed{C}$						◪			YES
FLIGHT UNIT	E	AMES PROTON ANALYZER							☒							NO
FLIGHT UNIT	F	OPTICAL ASPECT								◫						NO
FLIGHT UNIT	G	THERMISTER (BATTERY A)								◫						NO
FLIGHT UNIT	H	ION-ELECTRON DETECTOR											$\boxed{P}$			YES

			19 JUNE	20 JUNE	21 JUNE	22 JUNE	23 JUNE	23 JUNE	24 JUNE	25 JUNE	27 JUNE	29 JUNE	1 JULY	3 JULY	8 JULY	
FLIGHT SPARE	a	VIBRATION SHAKER		◫												NO
FLIGHT SPARE	b	GM COSMIC RAY TELESCOPE			◪						$\boxed{C}$					
FLIGHT SPARE	c	ION-ELECTRON DETECTOR				$\boxed{P}$							◪			YES
FLIGHT SPARE	d	TRANSMITTER						◧	$\boxed{R}$							NO
FLIGHT SPARE	e	BATTERIES								◧						NO
FLIGHT SPARE	f	PROGRAM SWITCH									$\boxed{M}$					NO
FLIGHT SPARE	g	DOUBLE TELESCOPE												◫		NO

IF THE SUBASSEMBLY MALFUNCTION (OR OTHER UNSATISFACTORY PERFORMANCE) OCCURRED IN FLIGHT, IT COULD HAVE CAUSED:

- ■ A MISSION FAILURE
- ◧ A PARTIAL MISSION FAILURE BUT ONLY WITH AN IMPROBABLE SIMULTANEOUS COMBINATION OF CIRCUMSTANCES.
- ◪ A LOSS OF DATA FROM THE AFFECTED EXPERIMENT

OTHER PROBLEMS ENCOUNTERED ARE DENOTED BY:

- $\boxed{P}$ QUESTIONABLE SUBASSEMBLY OPERATION.
- ▬ MARGINAL SUBASSEMBLY OPERATION.
- ☒ PROCEDURAL FAILURE.
- ◫ SPECIAL PROBLEMS.
- $\boxed{R}$ SUBASSEMBLY REPAIRED.
- $\boxed{C}$ SUBASSEMBLY CHANGED (REPLACED).
- $\boxed{M}$ SUBASSEMBLY MODIFIED.
- $\boxed{D}$ SUBASSEMBLY REDESIGNED.

Part 1

S-3 FLIGHT SPACECRAFT TESTING SUMMARY

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
A-2	5/27/61	Regulator Converter	26V AC square wave frequency out of tolerance.	Prior to vibration tests.	Unit replaced.	Replacement of converter was delayed until this date due to the lack of another unit.
B-4	5/29/61	RF Choke Circuit	One of the chokes was burned out when the battery-charging connector was unsoldered.	Thermal-vacuum pre-test set-up and checkout.	Circuit repaired.	Operational mishap.
C-4	5/30/61	Double Telescope	Telescope non-operative; one of the photomultiplier tubes burned out.	Pre-test checkout.	Flight spare telescope installed in spacecraft after having been satisfactorily vibration tested.	The possibility exists that this problem may have been due to a faulty power supply. Information not available at this time to establish cause of malfunction.
D-5	6/1/61	GM Telescope	Telescope malfunctioned; power supply burned out.	Chamber evacuation; chamber pressure approximately 1.4×10^{-4} mm Hg.	Flight spare unit successfully vibration tested, but failed in a separate vacuum test (6/1/61). Unit repaired but failed again; repaired and reinstalled in spacecraft.	See below.
D-11	6/10/61	GM Telescope	Coincidence readout lost. One of the GM tubes apparently malfunctioned.	150 Solar Aspect exposure.	Test continued.	Separate testing was conducted on several telescopes after Flight Spacecraft acceptance tests. Eventually GM Telescope, Ser. #3 was installed in the Flight Spacecraft. This telescope has been success-vibrated (separately) and passed thermal-vacuum testing in the Flight Spare spacecraft.

S-3 FLIGHT SPACECRAFT TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
E-7	6/3/61	Ames Proton Analyzer	The high voltage probe had been sheared off when the top cover was removed on 6/2/61.	Pre-test set-up and checkout.	None.	Monitoring of Ames experiment done via the telemetry only because of this incident.
F-8	6/4/61	Optical Aspect	Optical aspect appeared to be malfunctioning.	30° Solar Aspect exposure; initial checkout (−20°C on transmitter).	Test was halted and a new Aspect card installed; further investigation revealed that the external power supply to the exciter lamp was faulty. A new power supply was substituted and the vacuum test restarted after installation of the original Flight Aspect card.	--
G-8	6/4/61	Thermistor in Battery Pack "A"	The thermistor was indicating a temperature of +30°C via the telemetry. Indications were that the Battery temperature was about 0°C.	30° Solar Aspect exposure; initial checkout (−20°C on transmitter).	Test was halted because of F-8 above; thermocouples were installed on the battery to check calibration of the thermistor in subsequent tests.	Thermistor replaced 6/22/61.
H-11	6/9/61	Ion-Electron Detector	The telemetry indicated an "extra position" between positions 11 and 12 of the wheel. Probably requires an extra pulse to move wheel from position 11 to position 12.	150° Solar Aspect exposure.	None.	During final calibration of the unit on 7/7/61, the photomultiplier tube failed. Consequently, Acceptance Testing of this unit with a new tube was required and was completed successfully.

Part 2

S-3 FLIGHT SPARE TESTING SUMMARY

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
a-2	6/20/61	Vibration Shaker	The magnetic field of the Calidyne 177A disrupted the operation of the payload. Encoder oscillator periods were shifted, some experiments read out zeros.	Vibration.	Attempts were made to reduce the magnetic field, but were unsuccessful. Vibration tests were run despite the interference from the shaker. The spacecraft was checked out after each vibration exposure, in addition to monitoring operational parameters during the exposure.	--
b-3	6/21/61	GM Telescope	Telescope was non-operative after the random-thrust test. Operation intermittent during subsequent vibration tests.	Vibration.	After the random-thrust test, the top cover was removed; and all connectors to the Telescope were found to be secure. Unit removed from spacecraft after completion of vibration testing.	--
b-9	6/27/61	GM Telescope	No GM Telescope in Flight Spare spacecraft.	Third pre-test set-up and checkout.	GM Telescope (Ser. #3) installed in system after having successfully passed vibration tests.	This unit eventually was installed in the Flight Spacecraft after successfully completing the entire series of Flight Spare spacecraft thermal-vacuum exposures.

S-3 FLIGHT SPARE TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
b-9 (cont.)						Another Telescope (#5) was installed in the Flight Spare spacecraft after having passed (sub-assembly) vibration and thermal-vacuum tests. This Telescope was vibrated at Design Qualification levels.
c-4	6/22/61	Ion-Electron Detector	I&E electrometer yielded a readout with no input.	First pre-test set-up and checkout.	This condition was believed to have been a result of excessive humidity.	Unit functioned properly after evacuation of chamber began.
c-11	7/1/61	Ion-Electron Detector	During the test, the I&E absorber wheel stuck in one position. Operation was intermittent, followed by normal operation. Normal operation during cold soak test.	150° Solar Aspect exposure.	Test continued. At the completion of the thermal-vacuum testing, it was noticed that the I&E package had a rattling internal sound. Investigation revealed that one of the four screws used to mount the wheel's motor had worked loose (apparently a result of vibration). The loose screw had damaged the phosphor coating on the photomultiplier tube.	Repair and retest of this unit was necessitated.

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S-3 FLIGHT SPARE TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
d-6 d-7	6/23/61	Transmitter	The Transmitter would not load down properly; output power was zero.	30° Solar Aspect; initial checkout (−20°C on Transmitter).	Test halted; Transmitter functioned properly at room temperature; unit removed from spacecraft for repair. The 2N1141 amplification stage was misaligned. Transmitter retuned.	
e-8 f-9	6/25/61 6/27/61	Batteries Program Switch	The Batteries would not operate the spacecraft.	30° Solar Aspect; initial checkout (−20°C on Transmitter).	The test was halted; Batteries removed; lab tests on Batteries at −10°C duplicated conditions; investigation showed that at −10°C, the operating voltage plateau of the Batteries was less than the "under-voltage lockout." The Program Switch was adjusted so that under-voltage lockout would occur at 12.0V instead of the former 12.8V.	Identical problem appeared during subsequent vacuum test. Lockout had not been properly adjusted to 12.0V (actual value 12.5V). This value of 12.5V was still not low enough to permit operation of the spacecraft.

S-3 FLIGHT SPARE TESTING SUMMARY (Cont'd.)

KEY	DATE	ITEM	DESCRIPTION OF PROBLEM	ENVIRONMENT	ACTION TAKEN	REMARKS
g-12	7/4/61	Double Telescope	The current meter monitoring the 6.5V regulated supply pegged, thereby indicating a short circuit.	Cold soak (−10°C).	The meter was turned off. Test continued. After the completion of the cold soak, the chamber was opened, and it was found that the 6.5V line from the spacecraft to the chamber interface was shorted to the chamber wall.	Since turning the Double Telescope switch off prevented a checkout of the Single-Crystal Detector, the Pulse Height Analyzer, as well as the Double Telescope, it was decided that an overnight −10°C vacuum check would be conducted to verify operation of these units. Test completed satisfactorily.

APPENDIX M

TEST REPORT

FLIGHT SPACECRAFT - FLIGHT SPARE SPACECRAFT

VIBRATION TEST

APPENDIX M: TEST REPORT FLIGHT SPACECRAFT AND FLIGHT SPARE SPACECRAFT - VIBRATION TESTS

1. SUMMARY

The Flight Spacecraft and the Flight Spare Spacecraft were subjected to vibration tests starting May 27, 1961 and June 20, 1961, respectively.

The tests were conducted in accordance with the Acceptance Test Program for Flight Systems of the S-3 Energetic Particles Satellite.*

The Flight Spacecraft successfully completed all of the vibration tests.

The Flight Spare Spacecraft successfully completed all of the vibration tests with the exception of the GM Telescope.

2. TEST RESULTS

Flight Spacecraft:

The Flight Spacecraft did not experience any malfunctions and/or discrepancies during the prescribed vibration tests.

Data recorded during the Combustion Resonance (600 cps) thrust axis exposure indicated that the input acceleration to the spacecraft was ±56 g from 550 cps to 650 cps and that the associated force varied from ±250 pounds at 550 cps to ~ ±125 pounds at 650 cps.

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* Appendix E

Flight Spare Spacecraft:

The GM Telescope of the Flight Spare Spacecraft malfunctioned during the thrust axis - random vibration exposure.

No malfunctions of any other spacecraft subassemblies were experienced during the prescribed tests.

APPENDIX N

TEST REPORT

FLIGHT SPACECRAFT - FLIGHT SPARE SPACECRAFT

THERMAL-VACUUM TESTS

APPENDIX N: TEST REPORT FLIGHT SPACECRAFT - FLIGHT SPARE SPACECRAFT - THERMAL VACUUM TESTS

1. SUMMARY

The thermal-vacuum test of the Flight Spacecraft and the Flight Spare Spacecraft began on May 29, 1961 and June 21, 1961, respectively.

The tests were conducted in accordance with the Acceptance Test Program for Flight Systems of the S-3 Energetic Particles Satellite.*

The following exposures were conducted in the order listed:

1. 30° Solar Aspect
2. 150° Solar Aspect
3. low temperature (-10°C vacuum soak.)

Exposure 1 simulated the space environment if the Sun was at an angle of 30° to the longitudinal (spin) axis of the spacecraft and illuminating the top cover of the instrument compartment. Exposure 2 simulated the space environment if the sun was at an angle of 150° to the longitudinal axis of the spacecraft and illuminating the bottom covers of the spacecraft. Exposure 3 is a conventional test during which the spacecraft was "soaked" at a uniform chamber temperature.

The aspect angles for exposures 1 and 2 differ from those employed during Design Qualification testing because of revisions to the spacecraft thermal design. The high temperature soak was eliminated from the Acceptance Tests since the temperature would not exceed temperatures attained in the Solar Aspect exposures.

No major difficulties were encountered during testing of the Flight Spacecraft although malfunctions occurred in the Double Telescope and the GM Telescope. Because of these malfunctions, Acceptance Testing (on a subassembly level) was required of each of these items.

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* See Appendix E

During testing of the Flight Spare Spacecraft, difficulties were encountered with the Transmitter, Batteries and the Ion-Electron Detector. The problems with the Transmitter and Batteries were satisfactorily resolved during the tests but subassembly Acceptance Tests were required of the Ion-Electron Detector.

2. TEST CONDITIONS

Test Set-Up:

Figure N-1 indicates the mounting configuration of the spacecraft during the thermal-vacuum exposures. The light rings were utilized to establish the thermal gradient conditions for the 30° and 150° Solar Aspect exposures.

During the test, four "live" Solar Paddles were tested with the Flight Spacecraft and two with the Flight Spare Spacecraft. Provisions were made to illuminate each of the Paddles during the exposures.

A single lamp was positioned to control the heating of the Transmitter compartment and another to control the temperature of the Magnetometer Sensor. A third lamp illuminated the Solar Cell Damage Experiment.

Test Parameters:*

During the 30° Solar Aspect exposure, the temperature of the Transmitter was decreased to -20°C while the instrument compartment was maintained (using the lamps) above -5°C. After verification of the starting capabilities of the Transmitter (converter) at -20°C, the temperature of the instrument compartment was increased until the temperature of Battery "B" reached +32°C. The Transmitter attained a temperature of +3°C during this process. The spacecraft was then operated for a period of 48 hours.

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* See Tables N-1 and N-2

During the 150° Solar Aspect exposure, the temperature of the Transmitter was stabilized at +35°C, the Magnetometer (station 1) at -10°C. The instrument compartment attained an average temperature of approximately -5°C during this process. With these initial conditions established, the spacecraft was operated for a period of 48 hours.

The temperature of the chamber wall was maintained at approximately -50°C throughout each Solar Aspect exposure.

The third exposure was a 48 hour low temperature (-10°C) Vacuum Soak.

During the exposures, the following minimum chamber pressures were attained:

Exposure	Flight Spacecraft	Flight Spare Spacecraft
30° Solar Aspect	1.5×10^{-6} mm Hg	3.4×10^{-6} mm Hg
150° Solar Aspect	2.9×10^{-7} mm Hg	5.5×10^{-7} mm Hg
-10°C Vacuum Soak	5.6×10^{-7} mm Hg	3.5×10^{-7} mm Hg

3. TEST RESULTS - FLIGHT SPACECRAFT:

Prior to the test, during checkout operations, it was discovered that the Double Telescope was non-operative. One of the two photomultiplier tubes had burned out. A spare Double Telescope was tested successfully at Flight Acceptance vibration levels, and installed in the Flight Spacecraft.

During chamber evacuation, at approximately 1.4×10^{-4} mm Hg, the GM Telescope failed. The chamber was vented and the unit removed from the spacecraft. The failure was due to inadequate potting of the High Voltage Power Supply thus causing arcing and corona discharge.

A spare GM Telescope was tested successfully at Flight Acceptance vibration levels and exposed to a brief vacuum check prior to installation in the spacecraft. The vacuum check again produced a failure in the telescope. This was repaired and rechecked, but it failed once again.

Attempts to achieve an acceptable GM Telescope were suspended pending modifications to the design by the experimenter.

The spare GM Telescope was repaired once again, installed in the spacecraft, and the test restarted.

Design Qualification and Flight Acceptance tests of the GM Telescope were performed at a later date.*

An apparent malfunction of the Optical Aspect was observed at the start of the 30° Solar Aspect exposure. Also, the thermister located within the Battery"A" indicated unrealistic temperatures.

The chamber was vented and the problem with the Optical Aspect was traced to a faulty external power supply for the exciter lamp. Additional thermocouples were installed on the Battery pack to monitor the temperature during the test. The thermistor was replaced at a later date.

One of the geiger tubes in the GM Telescope malfunctioned during the 150° Solar Aspect exposure. The test was not halted to repair this difficulty since separate testing of the telescope would be performed at a later date.

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* See Appendix Q

4. TEST RESULTS - FLIGHT SPARE SPACECRAFT:

In the interim between thermal-vacuum testing of the Flight Spacecraft and the Flight Spare Spacecraft, a redesigned GM Telescope had successfully completed sub-assembly Design Qualification Tests and a second telescope had successfully completed Flight Acceptance vibration tests. Accordingly this latter telescope (Ser.#3), designated "Flight Unit," was installed in the Flight Spare Spacecraft for thermal-vacuum tests.

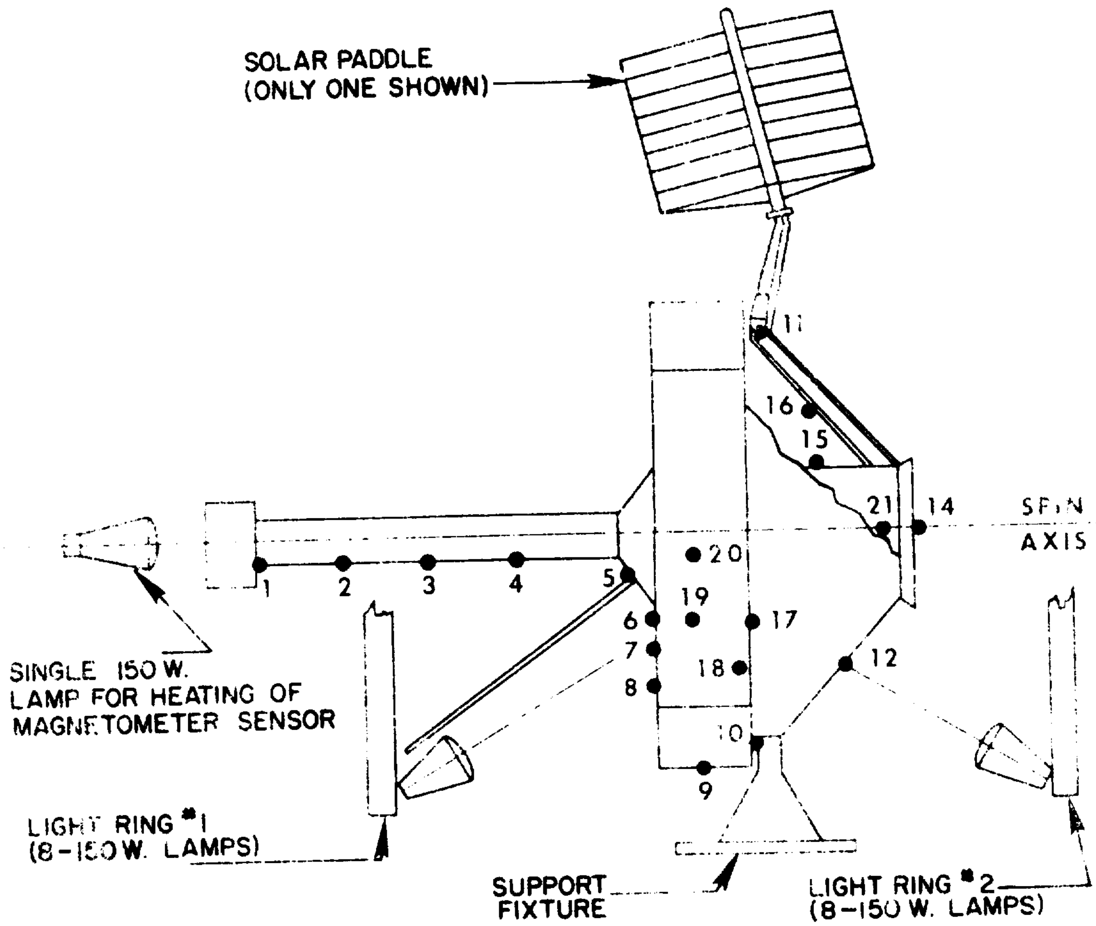
The initial checkout of the 30° Solar Aspect exposure revealed that the Transmitter (temperature -20°C) would not operate properly. The chamber was vented and the Transmitter removed from the spacecraft. It was determined that the amplifier (2N1141) was misaligned. The unit was tuned for proper operation.

The initial checkout during the restarted 30° Solar Aspect exposure yielded the fact that the spacecraft could not be operated on internal power. In addition, it appeared that the Batteries (temperature -10°C) would not accept a charge.

The test was halted and the Battery Packs were removed from the spacecraft. Subsequent checks revealed that the operating voltage plateau of the Batteries was less than the under voltage lockout point. (The Programs Switch senses the Battery voltage and, if this voltage is less than +12.8 v, will de-energize the spacecraft for an eight hour recycle period.) The Program Switch was adjusted so that the under voltage lock out would occur at +12.0 v and thus be compatible with the characteristics of the Batteries at -10°C.

The next attempt to conduct the 30° Solar Aspect exposure indicated once again, that the spacecraft would not operate on internal power. The under voltage sensing element of the Program Switch had not been readjusted with sufficient accuracy and was measured to operate at +12.5 v - still higher than the operating plateau of the Batteries. Readjustment of the setting to 12.0 v was accomplished after completion of the Thermal-Vacuum test.

The Ion-Electron Detector absorber wheel did not operate properly during the 150° Solar Aspect Exposure. The wheel was stuck in on position for some time but later operated intermittently. Normal operation was observed during the low temperature vacuum soak. At the conclusion of the test, it was determined that one of the four screws used to mount the motor had worked loose (apparently a result of the vibration test). The loose screw had damaged the phosphor coating on the photomultiplier tube. Because of the physical dimensions of the unit, repair was not possible and it was necessary to modify the design of this particular detector. Acceptance tests of the IR-Electron Detector (vibration and thermal-vacuum) were conducted on a subassembly level.



NOTE: Numbers refer to stations for which temperatures have been calculated by the Thermal Systems Branch. Skin temperature locations are indicated by dots attached to a line.

Figure N-1. Setup for the Thermal-Vacuum Test of the Flight Spacecraft and Flight Spare Spacecraft

TABLE N-1

Summary of Thermal Data
Flight Spacecraft

(degrees Centigrade)

LOCATION	30° SOLAR ASPECT			150° SOLAR ASPECT			-10°C VACUUM SOAK		
	B	A	ΔT	B	A	ΔT	B	A	ΔT
Telemetry Encoder	43.0	49.5	6.5	-4.0	-0.5	3.5	-11.0	-4.0	7.0
Regulator Converter	30.0	45.0	15.0	-4.0	9.0	13.0	-12.0	9.0	21.0
Battery Pack "A"	27.0	36.0	9.0	0	5.0	5.0	-10.0	3.0	13.0
Battery Pack "B"	33.0	40.5	7.5	1.5	5.5	4.0	-11.0	2.0	13.0
Pulse Height Analyzer	43.5	49.5	6.0	-5.0	-0.5	4.5	-11.5	-3.0	8.5
Power Transistor (Side)	6.0	12.0	6.0	18.5	38.5	20.0	-11.5	-4.0	7.5
Power Resistor (Side)	6.0	12.0	6.0	20.0	40.0	20.0	-11.0	-4.0	7.0
Power Transistor (Bottom)	10.0	17.0	7.0	19.5	31.0	11.5	-11.5	-3.0	8.5
Power Resistor (Bottom)	8.0	15.5	7.5	23.0	35.5	12.5	-11.0	-2.5	8.5
Transmitter	3.0	17.5	14.5	35.0	46.5	11.5	-11.0	5.0	16.0
Chamber Wall	-50.0	-48.0	-2.0	-54.5	-55.5	-1.0	-11.0	-8.0	3.0

TABLE N-2

Summary of Thermal Data
Flight Spare Spacecraft

(degrees Centigrade)

LOCATION	30° SOLAR ASPECT			150° SOLAR ASPECT			-10°C VACUUM SOAK		
	B	A	ΔT	B	A	ΔT	B	A	ΔT
Telemetry Encoder	40.0	51.5	11.5	-6.5	-3.5	3.0	-11.0	-4.5	6.5
Regulator Converter	30.0	43.0	13.0	-7.0	6.5	13.5	-11.0	5.5	16.5
Battery Pack "A"	33.5	44.0	10.5	-4.5	1.0	5.5	-11.0	-2.0	9.0
Battery Pack "B"	32.0	35.0	3.0	-4.5	-0.5	5.0	-11.5	-4.5	7.0
Pulse Height Analyzer	44.5	49.0	4.5	-7.0	-2.5	4.5	-11.5	-4.0	7.5
Power Transistor (Side)	4.0	29.0	25.0	18.0	39.0	21.0	-10.0	7.5	17.5
Power Resistor (Side)	4.0	27.5	23.5	18.0	38.0	20.0	-10.0	6.0	16.0
Power Transistor (Bottom)	5.0	24.0	19.0	15.0	30.0	15.0	-10.0	4.5	14.5
Power Resistor (Bottom)	5.0	22.0	17.0	16.0	30.0	14.0	-10.0	3.0	13.0
Transmitter	-0.5	16.0	16.5	35.0	45.0	10.0	-10.5	4.5	14.5
Chamber Wall	-56.0	-55.5	0.5	-58.0	-58.5	-0.5	-11.0	-10.0	1.0

Key:
B = Before spacecraft operation, temperatures stabilized.
A = After spacecraft operation, temperatures stabilized.
ΔT = (A-B) Temperature rise due to spacecraft operation.

APPENDIX P

BALANCE REPORT

APPENDIX P: BALANCE REPORT*

1. SUMMARY

Adequate balance of the S-3 Spacecraft was necessary to insure spin stability of the satellite during the launch phase and in orbital flight.

Available facilities did not permit dynamic balancing of the spacecraft in its orbital configuration (i. e., Solar Paddles erected). Therefore, the spacecraft was balanced in the launch configuration (Solar Paddles folded) and methods and procedures developed to permit the calculation of the degradation induced by erecting the paddles.

Although the Flight Spacecraft was precisely balanced prior to shipment to the launch site, substitution of electronic subassemblies within the spacecraft later became necessary and required analytical monitoring of the degradation of the balance condition if re-balance operations were to be avoided.

Satisfactory stabilization of Explorer XII in orbit (undetectable precession) justifies the conclusion that the balancing techniques and analytical methods and procedures utilized represent successful solutions to the problem of insuring adequate spin stability for spacecraft of this type.

The following summary of the balance data ** represents the residual unbalance of the Flight Spacecraft at the times indicated:

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* This report is based on T&E Memorandum Report 621-5 by W. E. Lang, Mechanical Test Section, Test and Evaluation Division.

** The balance specifications (allowable unbalance) for the S-3 were:

a. Launch configuration: 188 gram-in, static, 13,400 gram-in^2, dynamic

b. Orbital configuration: $\leq 1°$ precession angle (i.e., 15,600 gram-in^2 for the S-3)

1. Final Flight Balance at GSFC (July 8, 1961):

   a. Launch configuration: 33 gm-in, static,
      193 $gm\text{-}in^2$, dynamic

   b. Orbital configuration: 1600 $gm\text{-}in^2$, dynamic

2. After substitution of subassemblies at Cape Canaveral:

   a. Launch configuration: 306 gm-in, static
      785 $gm\text{-}in^2$, dynamic

   b. Orbital configuration: 1450 $gm\text{-}in^2$, dynamic

3. After rearranging Solar Paddles to improve balance condition:

   a. Launch configuration: 69 gm-in, static,
      4925 $gm\text{-}in^2$, dynamic

   b. Orbital configuration: 4870 $gm\text{-}in^2$, dynamic

4. After reviewing of the despin circuitry, removal of the spacecraft from the Launch Vehicle, and final balance of spacecraft/X-248 combination:

   a. Launch configuration: 176 gm-in, static,
      4925 $gm\text{-}in^2$, dynamic

   b. Orbital configuration: 4870 $gm\text{-}in^2$ ($1/6^\circ$- precession angle)

2. INTRODUCTION

The balance requirements for the S-3 were:

a. that the spacecraft, in its launch configuration be statically and dynamically balanced within limits imposed by Delta launch vehicle.

b. that the composite assembly (comprising the ABL X-248 rocket motor, the flight spin table base and support skirt, the separation device, and the spacecraft installed in launch configuration) be statically and dynamically balanced within limits imposed by the launch vehicle.

c. that the spacecraft in its orbital configuration (i.e. with Solar Paddles erected be statically and dynamically balanced after separation from the expended X-248 motor within limits imposed by allowable orbital precession.

Requirement (a) is imposed to minimize addition of weight to the composite assembly to meet requirement (b). Also there is an implied limitation on the unbalance induced by change of spacecraft configuration, since requirements (a) <u>and</u> (c) must be satisfied.

The spacecraft static balance tolerance is expressed in terms of displacement of the payload center-of-gravity from its geometric axis and therefore depends on the spacecraft weight. Static balance may be considered to be a launch phase requirement since the maximum center-of-gravity displacement specified for launch would have neglible effect on orbital stability.

Spacecraft dynamic balance restraints imposed by the launch vehicle are expressed in terms of the spin-axis moment-of-inertia of the composite assembly and, therefore, depend on the spacecraft inertia.

The stability of the spacecraft orbital configuration depends on the angle between the principal axis and the spin axis. The spin axis is, within alignment error, the geometric axis, or centerline of the spacecraft. This angle is a function of dynamic unbalance and of the ratio between the spin-axis and pitch-axis moments-of-inertia of the spacecraft. Therefore, the orbital configuration dynamic balance tolerance is determined by the allowable precession in orbit and by the inertial characteristics of the spacecraft. A spin-to-pitch inertial ratio close to 1.0 implies stringent dynamic balancing restraints for orbital stability (for the S-3 the ratio was approximately 1.3 and the allowable dynamic unbalance for acceptable orbital stability (1° precession angle) was relatively large).

3. <u>BALANCING THE S-3 SATELLITE</u>:

It was apparent that the four Solar Paddles would significantly affect the balance of both the launch (paddles folded) and orbital (paddles extended) configurations. Available balancing equipment could not be used to balance the orbital configuration because of the wind mill effect which would be created by spinning the spacecraft with the paddles extended. It was decided to balance the spacecraft with the paddles folded (i.e. in launch configuration) and secured by a spider fixture, and to determine the <u>change</u> of balance which would result from erection of the paddles into orbital configuration.

Since the paddles were nominally of equal weight and symmetrically located, the unbalance induced by paddle erection was due to small deviations from nominal weight and location. It was also necessary to consider the shift of the center-of-gravity of the entire spacecraft as a result of the extension of the paddles (this was about 1 5/8").

The technique developed required two basic steps:

a. the weight and center-of-gravity location of each solar paddle were determined for both launch and orbital configurations. The locations were measured using a cylindrical coordinate system based on the spacecraft spin axis. This was done immediately after balancing with the spacecraft still installed on the balancing machine arbor. Coordinates used were radius from spin axis, axial displacement from spacecraft interface, and angular orientation.

b. the unbalance was computed as the vector sum of the mass moment of each paddle. The total unbalance induced by erection of the paddles was computed as the effect of their removal from their folded position, plus their effect when extended.

As the unbalance resulted from small differences between large quantities, accurate measurements and exact calculations were necessary. Use of graphical methods or a slide rule would not suffice. It is also essential that organization of data and calculation format include careful observance of appropriate sign conventions.

This technique was applied to the S-3 Structural Prototype (with simulated paddles) to develop appropriate measurement procedures and to evaluate the problem in terms of necessary quality control of paddle weights and locations.

As a partial check of the technique, the static unbalance induced in the S-3 package by paddle erection was checked experimentally. The result compared with the calculated unbalance within 20% magnitude and within 10° phase angle. As the unbalance was small (approximately 20 oz/in) this was considered acceptable correlation. Experimental verification of the dynamic unbalance computation was not possible. This emphasizes the need for extreme care and accuracy in measurement and calculation.

The results indicated that:

a. The dynamic unbalance induced by erection of the paddles was approximately 275 oz/in^2. This value corresponds to a total precession angle of approximately 1° for the S-3 configuration. This leads to the conclusion that quality control of location was adequate provided that weight variation of the actual Flight Solar Paddles did not greatly exceed that of the simulated paddles.

b. It was apparent from the calculations that for the S-3 configuration approximately 90% of the induced dynamic unbalance was caused by the two "Low Paddles" because of their axial displacement from the center-of-gravity of the spacecraft. Therefore, maximum quality control of the low paddles was recommended.

Balancing of the Prototype Unit was done with the simulated paddles folded. The open-paddle check operation was not done because the numerical result would be significant only for the Flight Spacecraft with Flight Solar Paddles.

Although this report is primarily concerned with analytical techniques, it should be noted that balancing the S-3 was not a routine operation. Difficulty was encountered because of the considerable overhang of the configuration with paddles folded, and because the correction planes to be used (the top and bottom of the octagonal instrument compartment) were only 5.5" apart.

For the final balance of the Flight Spacecraft and Flight Spare Spacecraft, at GSFC, a variation of the usual plane transfer technique was developed. Instead of determining unbalance in two planes outside the spacecraft and transferring to the spacecraft correction planes, the unbalance was determined in the outboard spacecraft correction plane and the plane of the spider-fixture which held the paddles, and a new graphical plane transfer method was used. Evaluation of this new technique will continue; for the S-3 Flight Spacecraft acceptably low residual unbalances were obtained in less time than that usually required.

In retrospect, it is believed that S-3 balancing operations could have been improved by using the balancing machine to balance the orbital configuration but with paddles removed, and computing the unbalance effects of the paddles for both erected and folded positions. This would have made the machine balancing easier and probably more accurate (by reducing the overhang), would have eliminated handling of the expensive and delicate Solar Paddles, and would have simplified the calculations because the negative mass concept of removal of the paddles from their initial position need not be considered.

4. MONITORING THE BALANCE OF THE S-3 AT THE LAUNCH SITE:

It was anticipated that replacement of defective electronic subassemblies might be required at the launch site. In the event such a situation did arise, a procedure was developed to maintain adequate knowledge of the balance condition of the spacecraft.

The methods developed recognized four problem areas:

a. A bookkeeping problem; it was necessary that any action affecting the mass distribution of the spacecraft be recorded in a form suitable for evaluation.

b. Determination of balance condition from the balance record; the mechanics of this calculation are the same as for determining the unbalance induced by erection of the Solar Paddles. The unbalance is determined by a vector summation of the unbalance effects of all the changes of mass distribution.

c. Estimating the accuracy of the calculated unbalance; while the calculation, if done carefully, is theoretically error free, the dimensional data obtained from the balance record would necessarily be approximate.

Two attempts were made to evaluate the overall accuracy of the balance monitoring effort. The first was based on a check of the balance of the S-3 Flight Spare Spacecraft on the balancing machine at GSFC after several subassembly changes had been made. The second was based on the correction required at the final composite balancing operation on the Douglas Aircraft Company facility at Cape Canaveral. The results were inconclusive because of non-repeatability of alignment and lack of sensitivity of the balancing machines. However, it can be concluded that the monitoring effort was sufficiently accurate for S-3 mission objectives.

d. Evaluation of the calculated unbalance and decision on appropriate corrective action; separate records were kept of the static balance of the launch configuration, the dynamic balance of the launch configuration, and the dynamic balance of the orbit configuration. The balance condition of the Flight Spacecraft was monitored through the prelaunch period. The records were kept in the form of vector charts to indicate the magnitude and phase of the unbalance in relation to a circle representing the maximum allowable unbalance. These charts permitted "quick look" evaluation of the balance condition. If the unbalance exceeded the maximum allowable limit, a correction based on the calculated unbalance could be made or a rebalance of the spacecraft would be required.

The balance monitoring effort began at the time of the final Flight Balance of the spacecraft at GSFC. The residual static and dynamic unbalance was determined and recorded on the Unbalance Vector Charts. The computed unbalance induced by solar paddle deployment was also recorded as a vector component of the Orbital Configuration dynamic unbalance. All subsequent mass changes were expressed as vector unbalance components, and recorded on the charts.

The Specifications (or allowable unbalance limits) for the S-3 were:

a. Launch configuration (spacecraft only), static unbalance not to exceed 188 gm-in.

b. Launch configuration (spacecraft only), dynamic unbalance not to exceed 13,400 gm-in^2.

c. Orbital configuration, dynamic unbalance not to exceed 15,600 gm-in^2 (based on the requirement that the precession angle should not exceed 1°).

Residual unbalances achieved by final balance of the Flight Spacecraft at GSFC were:

a. Launch configuration, residual static unbalance: 33 gm-in.

b. Launch configuration, residual dynamic unbalance: 193 gm-in^2.

c. Orbital configuration, residual dynamic unbalance: 1600 gm-in^2 (including calculated degradation due to paddle erection).

During the four week period between the final balance and the launch date, six mass distribution changes were recorded which significantly affected the spacecraft balance. A number of other activities were noted and evaluated as inconsequential and, in most cases, not amenable to numerical estimate. Examples of these were minor wiring changes and removal and replacement of subassemblies without change (frequently necessary for access to the other subassemblies).

Significant balance changes were induced by the replacement of experiment subassemblies and the addition of electrical circuit components. The net result of these changes was degradation of balance condition to:

a. Launch configuration, static unbalance: 306 gm-in. (exceeded spec. of 188 gm in)

b. Launch configuration, dynamic unbalance: 785 $gm\text{-}in^2$. (degraded but well within spec.)

c. Orbital configuration, dynamic unbalance: 1450 $gm\text{-}in.^2$ (degraded but well within spec.)

Balance changes were also introduced because of replacements of modules in the Solar Paddles, which changed the weight and center-of-gravity position of each affected paddle. It was decided to exploit the paddle mass changes to optimize the overall balance condition by rearrangement of the paddles. This required calculations based on the data secured by precise measurements of folded and erected paddle positions. Several arrangements were analyzed, and an arrangement chosen which would reduce the launch configuration static unbalance within specification limits without causing excessive dynamic unbalance during launch or orbit. The new arrangement of Solar Paddles resulted in the following balance condition:

a. Launch configuration, static unbalance: 69 gm-in, Orbit configuration, static unbalance: 278 gm-in.

b. Launch configuration, dynamic unbalance: 4925 $gm\text{-}in^2$.

c. Orbit configuration, dynamic unbalance: 4870 $gm\text{-}in^2$.

The orbit configuration-static unbalance was considered in order to evaluate the possibility of tip-off at separation. It was decided that the unbalance, corresponding to 0.007 inches center-of-gravity displacement, was not significant.

The last significant mass change resulted from re-wiring was done because of premature firing of the despin dimple motors during pre-launch operations.) Accurate evaluation of the unbalance resulting from this change was not possible, but a conservative estimate was made that the launch configuration static unbalance did not exceed 176 gm-in, and that dynamic balance was not appreciably affected.

After balancing of the spacecraft/X-248 assembly and installation on the launch vehicle, the spacecraft had to be removed from the vehicle in order to replace the Transmitter. This made it necessary to rebalance the composite assembly. This did not affect the balance monitoring operation, except to complicate the attempt to evaluate its accuracy by introducing another unknown " non repeatable alignment" error.

At the time of launch, the balance condition of the S-3 was calculated to be:

a. Launch configuration, static unbalance: $\simeq$ 176 gm-in.
b. Launch configuration, dynamic unbalance: 4925 gm-in.^{2}
c. Orbital configuration, dynamic unbalance: 4870 gm-in^2

These estimates agree with available flight data which indicate that initial orbital precession of about 10°, (possibly induced by yo-yo despin) dampened to about 1° during paddle deployment, and that no precession was detectable after 24 hours. 4870 gm in^2 dynamic unbalance for orbital configuration implies a precession of about 1/6° which is less than the resolution of the telemetered data.

5. CONCLUSIONS

Satisfactory flight performance justifies the conclusion that the balancing techniques developed served their ultimate purpose, and therefore represent successful solutions to typical problems. The method used to calculate unbalance due to Solar Paddle erection and the procedure used to monitor balance condition during prelaunch field operations represent successful application of the same general principles.

If any physical change affecting the mass distribution of the spacecraft takes place at the launch site, then monitoring of the balance condition is the only alternative to rebalancing the spacecraft.

Available facilities for spacecraft balancing at Cape Canaveral are less sensitive than those at GSFC, do not permit rapid evaluation of residual unbalance, and require mounting the spacecraft in an inverted position.

The monitoring procedures utilized for the S-3 guaranteed acceptable balance condition and permitted the rearrangement of the Solar Paddles which restored adequate balance with no weight penalty.

It is noteworthy that the purpose of the balancing restraints imposed on the spacecraft by the launch vehicle is to limit the amount of ballast necessary to balance the composite assembly.

APPENDIX Q

SUBASSEMBLY TESTS

APPENDIX Q: SUBASSEMBLY TESTS

The reason for conducting tests at a subassembly level arose from three distinct program requirements:

1. Pre-system tests
2. Retest of an item after unsatisfactory performance during a system-level test
3. Acceptance tests of a second set of spares

"Pre-system tests" of subassemblies prior to integration into a spacecraft system) were intended to eliminate an excessive number of failures at the system level.

"Re-tests" of individual subassemblies (which had failed at the system level) were necessary so that retest of the entire spacecraft could be avoided.

Acceptance Tests of "Spare 2" subassemblies were necessary to provide an additional set of "flight worthy" packages in the event trouble developed at the launch site with the Flight or Flight Spare units.

The following tables include all of the subassembly tests conducted during the S-3 test program. The temperature and thermal-vacuum test limits are indicated in the tables while the vibration test levels, when different from standard Delta spacecraft input vibrations, are indicated in the Notes following the tables.

LOG OF SUBASSEMBLY TESTS CONDUCTED

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks—Test Results
1	Transmitter	Proto.	Vib.	See Note 1	Oct. '60	Satisfactory
2	Transmitter	Proto.	Acc.	28 g Thrust Axis	21 Oct. '60	Satisfactory
3	Battery	Eng.	Vib.	See Note 2	17 Oct. '60	Satisfactory
4	Battery	Eng.	Acc.	28 g Thrust 18 g Transverse	20 Oct. '60	Satisfactory
5	GM Tube for GM Telescope	Eng.	Vib.	See Note 3	Nov. '60	Satisfactory
6	Command Receiver*	Proto.	Vib.	Std. Delta Design Qual. Levels	Jan. '61	Satisfactory
7	Command Receiver*	Proto.	Acc.	28 g Thrust Axis	4 Jan. '61	Satisfactory
8	PM Tubes (11) for Double Telescope	Flight & Spare	Vib.	Std. Delta Flt. Acceptance Levels	Feb. '61	Satisfactory
9	Despin Timers (8)	Ser. No. 1-5, 7, 9, 10	Acc.	Special Design Test	22 Mar. '61	7 of 8 Satisfactory
10	GM Telescope GM Telescope	Flt. Flt.	Vib. Vac.	See Note 4 See Note 4	1 June '61 1 June '61	Satisfactory Failed
11	High Voltage Power Supply	Spare	Vib.	Std. Delta Flt. Acceptance Levels	7 June '61	Satisfactory
12	GM Telescope	Proto. #2	Vib.	See Note 5	17 June '61	Satisfactory
13	GM Telescope	Proto. #2	Acc.	28 g Thrust Axis	16 June '61	Satisfactory
14	GM Telescope	Proto. #2	Vac.			Failed
15	GM Telescope	Flight	Vib.	See Note 6	17 June '61	Satisfactory
16	GM Telescope	Ser. No. 3 Flight	Vib.	See Note 6	27 June '61	Satisfactory

*Not part of final spacecraft design

LOG OF SUBASSEMBLY TESTS CONDUCTED (Cont'd.)

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks—Test Results
17	GM Telescope	Ser. No. 5 Proto. #3	Vib.	See Note 5	27 June '61	Satisfactory
18	GM Telescope	Ser. No. 4 Spare	Vib.	See Note 6	30 June '61	Satisfactory
19	GM Telescope	Ser. No. 1 Spare	Vib.	See Note 6	30 June '61	Satisfactory
20	GM Telescope	Ser. No. 5 Proto. #3	Temp Cycling	50°C, 48 hrs. -10°C to +50°C Four Cycles	2 July '61	Satisfactory
21	GM Telescope	#5	Th-Vac.	+40°C, 24 hrs. -10°C, 12 hrs. +40°C, 12 hrs.	6 July '61	Satisfactory
22	GM Telescope	#4	Th-Vac.	+40°C, 24 hrs. -10°C, 12 hrs. +40°C, 12 hrs.	6 July '61	Satisfactory
23	GM Telescope	#1	Th-Vac.	+40°C, 24 hrs. -10°C, 12 hrs. +40°C, 12 hrs.	6 July '61	Satisfactory
24	GM Telescope	#4	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Failed
25	GM Telescope	#4	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	19 July '61	Satisfactory
26	PM Tubes (3 units)	-	Vib.	See Note 7	31 Mar.'61	-
27	Detector	Flight	Vib.	See Note 8	11 July '61	Satisfactory
28	Detector	Flight	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Satisfactory
29	Detector	Spare 1	Vib.	See Note 9	13 July '61	Satisfactory
30	Detector	Spare 1	Vib.	See Note 9	25 July '61	Satisfactory

LOG OF SUBASSEMBLY TESTS CONDUCTED (Cont'd.)

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks—Test Results
31	Detector	Spare 1	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Failed — Diode short in photo tube
32	PM Tube	Spare 2	Vib.	See Note 10	28 July '61	Failed
33	PM Tube	Spare 3	Vib.	See Note 11	29 July '61	Satisfactory
34	Detector (with Spare 3 Tube)	Spare 1	Th-Vac.		31 July '61	Satisfactory
35	PM Tube	Spare 4	Vib.	See Note 11	2 Aug. '61	Satisfactory
36	PM Tube	Spare 5	Vib.	Std. Delta Flt. Acceptance Levels	3 Aug. '61	Failed during random-thrust — sine thrust not done
37	Detector (with Spare 4 Tube)	Spare 1	Th-Vac.	+40°C, 12 hrs. -10°C, 12 hrs.	7 Aug. '61	Satisfactory
38	PM Tube	Spare 6	Vib.	See Note 11	10 Aug. '61	Satisfactory
39	Regulator Converter	Spare 6352	Vib.	See Note 12	10 July '61	Satisfactory
40	Regulator Converter	Spare 6580	Vib.	See Note 12	10 July '61	Satisfactory
41	Regulator Converter	Spare 6581	Vib.	See Note 12	10 July '61	Satisfactory
42	Regulator Converter	Spare 6352	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Failed — unknown material leaked out during test
43	Regulator Converter	Spare 6580	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Failed — unknown material leaked out during test
44	Regulator Converter	Spare 6581	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Not operated during test. This, along with 6352 & 6580, returned to manufacturer.

LOG OF SUBASSEMBLY TESTS CONDUCTED (Cont'd.)

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks—Test Results
45	Regulator Converter	Spare 6584	Vib.	See Note 12	18 July '61	Satisfactory
46	Regulator Converter	Spare 6579	Vib.	See Note 12	18 July '61	Satisfactory
47	Regulator Converter	Spare 6583	Vib.	See Note 12	18 July '61	Satisfactory
48	Regulator Converter	Spare 6584	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	22 July '61	Satisfactory
49	Regulator Converter	Spare 6579	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	22 July '61	Satisfactory
50	Regulator Converter	Spare 6583	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	22 July '61	Not operated during test
51	Regulator Converter	Spare 6583	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Satisfactory
52	Power Transistors (3)	ZN1616 6039T	Vib.	Std. Delta Flt. Acceptance Levels — Thrust Axis Only	27 April '61	To be used in solar array voltage regulator
53	Battery "A"	Spare 2	Vib.	See Note 13	21 July '61	Satisfactory
54	Battery "B"	Spare 2	Vib.	See Note 14	21 July '61	Satisfactory
55	Battery "A"	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Satisfactory
56	Battery "B"	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Satisfactory

LOG OF SUBASSEMBLY TESTS CONDUCTED (Cont'd.)

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks— Test Results
57	Program Switch	Spare 2	Vib.	Std. Delta Flight Acceptance Levels	21 July '61	Satisfactory
58	Recycle Timer	Spare 2	Vib.	Std. Delta Flight Acceptance Levels	21 July '61	Satisfactory
59	Program Switch	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Failed — units did not operate properly throughout the test.
60	Recycle Timer	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	
61	Encoder Converter	Spare 2	Vib.	See Note 15	4 Aug. '61	Satisfactory
62	Optical Aspect Converter	Spare 2	Vib.	See Note 16	4 Aug. '61	Satisfactory
63	Encoder Converter	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 12 hrs.	7 Aug. '61	Satisfactory
64	Optical Aspect Converter	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 12 hrs.	7 Aug. '61	Satisfactory
65	Transmitter	Spare 2	Vib.	Std. Delta Flight Acceptance Levels	27 July '61	Satisfactory
66	Transmitter	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 12 hrs.	28 July '61	Satisfactory — this unit was flown in Explorer XII
67	Transmitter	(former) Flight	Vib.	Std. Delta Flight Acceptance Levels	3 Aug. '61	Satisfactory — unit had been removed from flight spacecraft because of accidental destruction.

LOG OF SUBASSEMBLY TESTS CONDUCTED (Cont'd.)

	Test Item	Model or Serial No.	Test Category	Test Levels	Date Completed	Remarks— Test Results
68	Transmitter	(former) Flight	Th-Vac.	+40°C, 12 hrs. -10°C, 12 hrs.	9 Aug. '61	Failed
69	Double Telescope	Spare — Replaces Flight Unit	Vib.	See Note 17	31 May '61	Satisfactory — this unit was flown in Explorer XII
70	Double Telescope	(Ser. #4) Spare 2	Vib.	See Note 18	21 July '61	Satisfactory
71	Double Telescope	(Ser. #4) Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Satisfactory
72	Pulse Height Analyzer	Spare 2	Vib.	See Note 19	10 July '61	Satisfactory
73	Pulse Height Analyzer	Spare 2	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	15 July '61	Failed
74	Pulse Height Analyzer	Spare 2	Th-Vac.	+40°C, 12 hrs. -10°C, 6 hrs.	28 July '61	Satisfactory
75	Single Crystal Detector	Spare 2 (Ser. No. 4)	Vib.	See Note 20	19 July '61	Satisfactory
76	Single Crystal Detector	Spare 2 (Ser. No. 4)	Th-Vac.	+40°C, 24 hrs. -10°C, 24 hrs.	19 July '61	Satisfactory

NOTES: VIBRATION LEVELS FOR S-3 SUBASSEMBLY TESTS

The levels utilized were based on transmissibility data* obtained during vibration tests of the S-3 Prototype Unit.**

In general, all tests included sinusoidal and random vibration in three orthogonal axes. The sinusoidal-thrust axis input vibration for individual subassemblies is indicated in table. The remaining five vibration exposures were conducted at standard Delta Design Qualification or Flight Acceptance Levels, unless otherwise indicated.

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* In applying the transmissibility data from the Prototype Unit to test levels for "hard table" subassembly tests, no attempt was made to determine harmonic improvements of the recorded data.

** See Appendix H

NOTES: VIBRATION LEVELS FOR S-3 SUBASSEMBLY TEST

NOTE	TEST ITEM	FREQUENCY (cps)	THRUST AXIS (± g's)	TRANSVERSE AXES (± g's)
1	Transmitter	5-50	2.1	0.8
		50-500	10.3	2.1
		500-2000	21.0	4.2
		2000-2400	50	28
2	Battery	5-50	2.1	0.8
		50-500	10.3	2.2
	(No random tests)	500-2000	21.2	4.2
		2000-3000	28	28
		3000-5000	15	28
3	GM Tube	5-50	5	0.8
		50-80	15	2.1
	(No random tests)	80-100	60	2.1
		100-160	20	2.1
		160-300	30	2.1
		300-500	10	2.1
		500-2000	21	4.2
		2000-3000	56	17
4	GM Telescope	10-50	1.5	Std.
		50-90	10	
		90-200	30	
		200-2000	5	
		2000-2500	35	
5	GM Telescope	10-50	2.2	Std.
		50-90	13	
		90-200	40	
		200-500	10	
		500-2000	3	
		2000-2500	55	
6	GM Telescope	10-50	1.5	Std.
		50-90	10	
		90-200	30	
		200-500	7	
		500-2000	2	
		2000-2500	35	

NOTES: VIBRATION LEVELS FOR S-3 SUBASSEMBLY TEST (Cont'd.)

NOTE	TEST ITEM	FREQUENCY (cps)	THRUST AXIS (± g's)	TRANSVERSE AXES (± g's)
7	PM Tube	5-60	2	Std.
		60-120	40	
		120-2000	15	
		2000-3000	20	
8	Ion Electron Detector	10-50	1.5	Std.
		50-80	9	
		80-150	54	
		150-300	7	
		300-500	2	
		500-2000	5	
		2000-3000	3	
9	Ion Electron Detector	10-50	1.5	Std.
		50-80	9	
		80-110	45	
		110-150	40	
		150-300	7	
		300-500	2	
		500-2000	5	
		2000-3000	3	
10	PM Tube	10-50	1.5	Std.
		50-80	9	
		80-150	42	
		150-300	7	
		300-500	2	
		500-2000	5	
		2000-3000	3	
11	PM Tube	10-50	1.5	Std.
		50-80	9	
		80-150	54	
		150-300	7	
		300-500	2	
		500-2000	5	
		2000-3000	3	

NOTES: VIBRATION LEVELS FOR S-3 SUBASSEMBLY TEST (Cont'd.)

NOTE	TEST ITEM	FREQUENCY (cps)	THRUST AXIS (± g's)	TRANSVERSE AXES (± g's)
12	Regulator Converter	10-50	1.5	Std.
		50-80	8	
		80-150	27	
		150-200	8	
		200-600	2	
		600-2000	5	
		2000-3000	13	
13	Battery "A"	5-50	1.5	Std.
		50-80	10	
		80-150	30	
		150-200	8	
		200-2000	5	
		2000-3000	8	
14	Battery "B"	5-50	1.5	Std.
		50-80	10	
		80-200	30	
		200-500	7	
		500-2000	2	
		2000-3000	25	
15	Encoder Converter	5-50	5	Std.
		50-100	35	
		100-3000	4	
16	Optical Aspect Converter	5-100	4	Std.
		100-500	6	
		500-2000	20	
		2000-3000	5	
17	Double Telescope (Random: .03g^2-cps; 20-1000 cps; 5.5 g-rms)	5-50	1.5	.6
		50-500	7.1	1.4
		500-1000	14	2.8
18	Double Telescope	5-50	1.5	Std.
		50-500	7.1	
		500-2000	14	
		2000-3000	36	

NOTES: VIBRATION LEVELS FOR S-3 SUBASSEMBLY TEST (Cont'd.)

NOTE	TEST ITEM	FREQUENCY (cps)	THRUST AXIS (± g's)	TRANSVERSE AXES (± g's)
19	Pulse Height	10-50	1.5	Std.
		50-90	8	
		90-150	35	
		150-2000	3	
		2000-3000	7	
20	Single Crystal Detector	10-50	1.5	Std.
		50-80	9	
		80-150	30	
		150-200	9	
		200-800	2	
		800-2000	6	
		2000-3000	15	

APPENDIX R

SPACECRAFT OPERATIONS AT THE AFMTC, CAPE CANAVERAL

PART I

OPERATION PLAN AND SCHEDULE

PART II

CHRONOLOGY OF EVENTS

PART 1: OPERATIONS PLAN AND SCHEDULE *

1.0

INTRODUCTION

1.1 S-3 Operations Plan

A general operations plan for the S-3 Project is being compiled by the Operations Division of the GSFC. Post-launch operations, launch communications, trajectory information, receiving station equipment and operation, spacecraft description, mission objectives, organization, etc., are described in that document and are not included here. Other working documents on the S-3 Project include a Project description, and test and evaluation manuals.

1.2 Document Function

This document entitled "S-3 Spacecraft Operations AMR," is to serve the following functions:

(a) To serve as a countdown guide for GSFC spacecraft personnel
(b) To inform the GSFC, Fields Project Branch on requirements and operations
(c) To provide a countdown from which spacecraft countdown milestones can be extracted for the Delta S-3 Countdown Manual

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* This document was prepared by P. G. Marcotte, Assistant Project Manager, S-3.

2.0

SPACECRAFT TASK SCHEDULE
F - 19 DAYS THROUGH F - 2 DAYS

TIME	TASK OR EVENT
F - 30 Days	(a) Delivery of Ground Receiving Station to Hanger AE.
F - 29 Days	(a) Delivery of Prototype Spacecraft to Hanger AE. (b) Checkout and setup ground station in Hanger AE. (c) Install Yagi antenna at Hanger AE.
F - 28 Days	(a) Checkout of Prototype Spacecraft.
F - 27 Days	(a) Prototype preliminary R.F. check from pad 17B to Hanger AE.
F - 26 Days	(a) Delivery of 500 microcurie test source to Hanger AE. (b) Delivery of Flight Unit Spacecraft to Hanger AE.
F - 25 Days	(a) Flight Unit - complete checkout of spacecraft (check all regulated voltages and performance parameters, etc.)
F - 24 Days	(a) Prototype R.F. antenna pattern measurements at Satellite Tracking Station.
F - 23 Days	(a) Flight Unit R.F. antenna pattern measurements at Satellite Tracking Station.
F - 22 Days	(a) Prototype preliminary R.F. test from DAC spin facility to Hanger AE.
F - 21 Days	(a) Flight Unit checkout of solar paddles individually. (b) Flight Unit test of solar cell damage experiment.

2.0 (continued)

TIME	TASK OR EVENT
F - 20 Days	(a) Flight Unit test of Solar Paddles while running spacecraft. Spacecraft mounted on motorized spin table. (b) Flight Unit - point I & E experiment at sun and check for increase in dark current. (c) Flight Unit - checkout of Optical Aspect on motorized spin table. (d) Flight Unit - install thermistor on Solar Paddle.
F - 19 Days	(a) Delivery of Flight Spare to Hanger AE.
F - 18 Days	(a) SUI Experiment exchange from Flight Spare to Flight Unit.
F - 17 Days	(a) SUI Flight Unit Calibration. (b) SUI Flight Spare Calibration.
F - 16 Days	(a) Flight Unit - Calibration of Double Telescope and RIDL package. (b) Flight Unit - Mu Meson check of Double Telescope. (c) Flight Spare - complete checkout of spacecraft (check all regulated voltages and performance parameters, etc.)
F - 15 Days	(a) Flight Unit - Calibration of Cosmic Ray Logic Box, GM Telescope and Single Crystal Detector. (b) Flight Spare - test of solar cell damage experiment. (c) Flight Spare - Point I & E experiment at sun and check for increase in dark current. (d) Flight Spare - check of Optical Aspect on motorized spin table.
F - 14 Days	(a) Flight Unit - Ames Proton Analyzer calibration. (b) Flight Spare - calibration of Double Telescope and RIDL package, Cosmic Ray Logic Box, GM Telescope, and Single Crystal Detector. (c) Flight Spare - Mu Meson check of Double Telescope.

TIME	TASK OR EVENT
F - 13 Days	(a) Prototype group - install Blockhouse Control Box and checkout. (b) Flight Spare - Ames Proton Analyzer calibration.
F - 12 Days	(a) Prototype - Mate to dummy 3rd stage at DAC spin facility. (b) Flight Unit - to DAC spin facility for fit check and attachment of solar paddle hold down brackets to 3rd stage motor.
F - 11 Days	(a) Prototype Spacecraft on stand. (b) Flight Unit - I & E Calibration. (c) Flight Unit - Magnetometer Check. (d) Flight Spare - to DAC spin facility for fit check to 3rd stage motor.
F - 10 Days	(a) R.F. COMPATIBILITY TESTS - PROTOTYPE Simulate Launch Countdown All stations manned All range instrumentation turned on Checkout Blockhouse Control Box Cable for pickup Measure R.F. signal strength at Hangar AE, and Satellite Tracking Station.
F - 9 Days	(a) Flight Unit - Install de-spin dimple motors. (b) Flight Unit - final assembly & mechanical checks. (c) Flight Unit - touch up spring seat thermal paint. (d) Flight Spare - I & E calibration. (e) Flight Spare - Magnetometer check.
F - 8 Days	(a) Flight Unit to DAC spin facility for final balance & alignment.
F - 7 Days	(a) Flight Unit at DAC spin facility. (b) Flight Unit - discharge and begin charge of batteries at end of day.

2.0 (continued)

TIME	TASK OR EVENT
F - 6 Days	(a) Flight Unit at DAC spin facility.
F - 5 Days	(a) Flight Unit mounted on stand.
F - 4 Days	(a) Flight Unit on stand - complete checkout.
F - 3 Days	(a) " " " " " "
F - 2 Days	(a) " " " " " "

3.0

SPACECRAFT TASK SCHEDULE
PREVIEW FOR F - 1 DAY AND F - 0 DAY

TIME		TASK OR EVENT
F - 1 Day		
T - 360 Min.	(a)	Flight Unit complete checkout according to Countdown Manual.
to	(b)	Connect battery charger and leave on.
	(c)	Review of receiving station status with
T - 270 Min.		GSFC Operations Control Center.
F - 0 Day		
T - 650 Min.	(a)	Remove stripcoat and touch up thermal coating.
to	(b)	Mechanical checks according to Countdown Manual.
	(c)	Optional spacecraft checkout period.
T - 385 Min.	(d)	Turn off and remove battery charger.
	(e)	Install Fairing.
T-235 to T-145 Min.	(f)	Spacecraft checkout according to Countdown Manual.
	(g)	Service tower removal.
T - 20 Min.	(h)	Final Spacecraft Turn On.
T - 0 Min.	(i)	LAUNCH

4.0

SPACECRAFT FLIGHT COUNTDOWN
FOR F - 1 DAY

TIME	TASK OR EVENT
T - 360 Min.	BEGIN SPACECRAFT CHECKS
	(1) VISUALLY INSPECT SPACECRAFT.
	(2) ATTACH SPACECRAFT TURN-ON AND MONITOR PANEL.
	(3) SET EXTERNAL PWR. SUPPLY TO 15.0 VOLTS.
	(4) RELEASE ANTENNAE.
	(5) TURN ON POWER SWITCH TO EXTERNAL POSITION.
	(6) ACTIVATE BLOCKHOUSE CONTROL TO TURN ON SPACECRAFT.
	(7) READ AND RECORD SPACECRAFT VOLTAGES AND CURRENTS.
	(8) READ OUT CHANNEL 2 ON THE TELEMETRY.
	(9) REMOVE ALL THREE COVER PLATES FROM THE OPTICAL ASPECT SENSOR AND VISUALLY INSPECT THE PRISM FACES.
	(10) REPLACE TOP AND BOTTOM ASPECT COVERS.
	(11) CHECK FOR COMPLETION OF ITEM #8.
	(12) APPLY FLASHLIGHT BEAM TO ASPECT SENSOR AND CHECK OUT PERFORMANCE ON TELEMETRY.
	(13) REPLACE FINAL ASPECT COVER.
	(14) PLACE 500 MICROCURIE SOURCE ON TOP DECK OF SPACECRAFT.
	(15) READ AND RECORD CHANNELS 3,4,5,6 and 11.
	(16) REMOVE RADIOACTIVE SOURCE FROM SPACECRAFT AND REPLACE IT IN THE LEAD CONTAINER.
	(17) ATTACH AMES H.V. PROBE TO SENSOR.
	(18) READ AND RECORD CHANNEL #7.
	(19) REMOVE AMES TEST BOX.
	(20) REMOVE SOLAR PADDLE SHIELD.

4.0 (continued)

(21) RELEASE SOLAR PADDLE ARMS.

(22) MONITOR CHANNELS 8,9,10 AND PERFORMANCE CHANNEL #7 FOR ONE COMPLETE ROTATION OF I & E ABSORBER WHEEL.

(23) APPLY FLASHLIGHT BEAM TO ONE APERTURE OF THE I & E SENSOR.

(24) READ OUT CHANNEL #8 FOR ONE COMPLETE ROTATION OF THE ABSORBER WHEEL.

(25) ALIGN ABSORBER WHEEL BY ALIGNMENT MARK AND TURN OFF ABSORBER WHEEL SWITCH ON THE METER PANEL.

(26) CONFIRM ALIGNMENT OF ABSORBER WHEEL FROM TELEMETRY.

(27) OBSERVE AND RECORD PERFORMANCE PARAMETER #1 AS THE PADDLES ARE LOWERED ONE BY ONE.

(28) TIE DOWN PADDLE ARMS.

(29) INSTALL SOLAR PADDLE SHIELD.

(30) INSTALL COIL SYSTEM OVER MAGNETOMETER SENSOR AND EXERCISE SENSOR.

(31) READ OUT CHANNELS 12,13 AND 14, REMOVE MAGNETOMETER COILS.

(32) TURN SPACECRAFT <u>OFF</u>.

(33) REMOVE TURN ON PLUG.

(34) SECURE ANTENNAE AGAINST MAGNETOMETER BOOM.

(35) INSTALL BATTERY CHARGER CABLE.

(36) TURN ON BATTERY CHARGER.

T - <u>270</u> Min. <u>END OF SPACECRAFT CHECKS.</u>

4.0 (continued)

SPACECRAFT FLIGHT COUNTDOWN
F - 0 DAY

T - 650 Min.	STRIP SPACECRAFT OF PROTECTIVE COATING.
	INSPECT THERMAL COATING (TOUCH UP AS NECESSARY)
	REMOVE TIMER SAFETYS (2 SCREWS)
	REMOVE TOP AND BOTTOM COVERS FROM OPTICAL ASPECT SENSOR.
	REMOVE APERTURE COVERS FROM I&E SENSOR.
	REMOVE SOLAR PADDLE SAFETY SHIELDS.
	CHECK YO-YO AND OTHER MECHANICAL & STRUCTURE HARDWARE.
to	INSPECT SINGLE XTAL DET. TO BE SURE THAT PROTECTIVE COVER HAS BEEN REMOVED.
	REMOVE PROTECTIVE COVER FROM SOLAR DAMAGE EXPERIMENT.
	CHECK TO SEE THAT ANTENNAE ARE FOLDED.
	TURN OFF AND REMOVE BATTERY CHARGER.
	MAKE FINAL VISUAL INSPECTION OF SPACECRAFT.
	INFORM TEST CONDUCTOR THAT SPACECRAFT CHECKS ARE COMPLETE AND THAT THE SPACECRAFT IS READY FOR FAIRING INSTALLATION.
T - 385 Min.	CONFIRM REMOVAL OF TOOLS TAKEN UP TO TOWER.

4.0 (continued)

SPACECRAFT FLIGHT COUNTDOWN
F - 0 DAY

T - 235 Min.	BEGIN SPACECRAFT CHECKS.
	RELEASE SPACECRAFT ANTENNAE.
	INSTALL TURN ON PLUG ON METER PANEL INTO THE SPACECRAFT.
	SET POWER SUPPLY TO 15.0 V.D.C.
	TURN MAIN POWER SWITCH TO EXTERNAL POSITION.
	EXERCISE BLOCKHOUSE CONTROL TO TURN PAYLOAD ON.
	PLACE 500 MICROCURIE RADIOACTIVE SOURCE ON THE TOP OF THE SPACECRAFT.
	READ AND RECORD SPACECRAFT VOLTAGES AND CURRENTS.
	READOUT TELEMETRY.
	REPORT WHEN TELEMETRY READOUT IS COMPLETE.
	TURN SPACECRAFT OFF.
to	WITHDRAW RADIOACTIVE SOURCE AND PLACE IT IN ITS PROTECTIVE CONTAINER.
	WITHDRAW METER PANEL TURN ON PLUG.
	INSTALL FLIGHT TURN ON PLUG (AND LOCKTIGHT).
	EXERCISE BLOCKHOUSE CONTROL TO TURN ON SPACECRAFT.
	ANALYZE TELEMETRY TO CONFIRM PROPER SPACECRAFT OPERATION. REPORT WHEN COMPLETE.
	EXERCISE BLOCKHOUSE CONTROL TO TURN OFF SPACECRAFT.
	CHECK TO SEE IF ANTENNAE ARE FOLDED PROPERLY AGAINST FAIRING.
	INFORM TEST CONDUCTOR THAT SPACECRAFT IS READY FOR TOWER REMOVAL.
T - 145 Min.	CONFIRM REMOVAL OF TOOLS TAKEN UP TO TOWER.

4.0 (continued)

SPACECRAFT TERMINAL COUNTDOWN
F - 0 DAY

TIME	TASK OR EVENT
T - 40 Min.	STATUS REPORT ON RECEIVING STATIONS FROM GSFC OPERATIONS CONTROL CENTER.
T - 35 Min.	TERMINAL COUNT BEGINS.
T - 20 Min.	TURN SPACECRAFT ON WITH BLOCKHOUSE CONTROL. MAKE COMPLETE CHECKOUT OF SPACECRAFT.
T - 10 Min.	STATUS REPORT ON SPACECRAFT TO PROJECT MANAGER.
T - 0	LAUNCH.

5.0

HOLD CRITERIA

5.1 Spacecraft Holds

(a) Any spacecraft malfunction on the Flight Day will cause a spacecraft hold.

(b) Any spacecraft malfunction on F-1 to F-0 days will cause a spacecraft hold unless there is sufficient time for installation and checkout of an alternate package.

(c) Any damage to, or change in, the spacecraft thermal coatings will cause a spacecraft hold.

(d) Any malfunction in operating the spacecraft from the blockhouse control will cause a spacecraft hold.

(e) Lack of an adequate spacecraft monitor due to R.F. interference, etc. will cause a spacecraft hold. (A spare spacecraft monitor set will be available).

5.2 Receiving Station Holds

A hold will be called if a primary receiving station (Joburg, Woomera, or Santiago) is not operational on the Flight Day or if equipment troubles appear near or during the terminal count.

5.3 Communication Holds

Loss of communication on operational channels between AMR and the NASA/GSFC, Operations Control Center will cause a hold.

Loss of communication between the NASA/GSFC Operations Control Center and the Joburg, South Africa Station will cause a hold.

6.0

COMMUNICATIONS

6.1 Spacecraft checks prior to T-1 Day

Telephone communication between the spacecraft laboratory, Cape Satellite Tracking Station, Blockhouse Position 48, and the gantry spacecraft level will be required for spacecraft checks, MOPS channels designated spacecraft No. 1 and spacecraft No. 2 will be used for these checks.

During spacecraft checks approximately 2.0 watts of R.F. power at 136.020 MC will be radiated.

6.2 Launch Communications

Spacecraft communication requirements during the launch phase will be designated following R.F. compatibility tests on F-10 days.

6.3 Communication with GSFC, Operations Control Center

Requirements are to be fixed at a later date in agreement with the Operations Plan. In addition to the Cape Teletype and Cape Doppler lines, two telephone lines will probably be used.

6.4 Post-Launch Telemetry Reception

A telemetry receiving and recording trailer will be installed at the Cape Minitrack Site for reception from:

(a) Launch to T + 1 hour.

(b) T + 8 hours to T + 24 hours,
(based on a nominal trajectory).

Predictions on satellite location vs. time will be provided Cape Minitrack from the GSFC, Operations Control Center.

6.5 R-F Silence Requirements

The radio frequency band 135.0 to 137.0 Megacycles is to be free from signals other than the spacecraft transmitter signal during the following periods:

(a) During spacecraft tests on F - 10 days, the day of the R-F compatibility check.

(b) During the spacecraft check on F - 4 days to F - 1 day.

(c) During all spacecraft checks on F - 0 days.

(d) Throughout the terminal countdown and from T - 0 minutes to T + 60 minutes.

(e) From T + 8 hours to T + 24 hours.

Additional requests for R-F silence may be made if found necessary during other test periods.

7.0

SPECIAL SERVICE REQUIREMENTS FOR GSFC, FIELD PROJECTS BRANCH

The following list contains items essential to the S-3 operation which are to be provided by the Fields Projects Branch. The list is primarily a reminder as these and other items have been previously discussed with the Fields Project Branch.

1. Space and power within the Cape Satellite Tracking Station for the spacecraft monitor function.

2. Space and power within the Spacecraft Laboratory for the spacecraft monitor function, checkout, storage, mechanical work, personnel, etc.

3. A Helmholtz coil in the Cape Satellite Tracking Station area for magnetic field checks.

4. A wooden platform for R-F power and pattern checks in the Cape Satellite Tracking Station area.

5. Space and power for the S-3 antenna, telemetry, and spare part trailer at the Cape Satellite Tracking Station.

6. Space for an antenna suitable for spacecraft monitoring at the Spacecraft Laboratory.

7. A blockhouse area with sufficient space for the blockhouse spacecraft control unit and two operators.

8. Electrician services for power hook-up to the telemetry trailer as required.

9. Telephone and Loudspeaker to the telemetry trailer --- loudspeaker for launch countdown.

PART 2: CHRONOLOGY OF EVENTS

SPACECRAFT PRELAUNCH OPERATIONS, CAPE CANAVERAL, FLORIDA

This report is a narrative discussion of the events which took place after the completion of the Test Program. The majority of the report is concerned with those events which took place during the Pre-launch Operations at the Atlantic Missile Range, Cape Canaveral.

It is not intended that this discussion serve as a detailed description of the various spacecraft and/or vehicle tasks which are necessary to the pre-launch operations (see Part 1, this Appendix) but rather a discussion of pre-launch problems and events of a general interest as well as those which may be of some value in estimating spacecraft dependability.

1. After the completion of Acceptance Tests, the Flight Spacecraft were taken to the Naval Ordnance Laboratory and the Fredicksburg, Va. magnetic test facility for magnetic field measurements and calibration of the magnetometer. During the course of one of these trips, a malfunction occurred in the Flight Spare Spacecraft which included the failure of the Program Switch and the Regulator Converter. No satisfactory explanation for this malfunction was apparent.

2. The Prototype Unit was shipped to Cape Canaveral on July 5 and was used for preliminary checkouts of the Ground Station, RF compatibility and other tasks at the Cape.

3. The Flight Spacecraft and the Flight Spare Spacecraft were shipped (via air) on July 10 and July 19, 1961 respectively. The Flight Spacecraft was transported to AMR without inclusion of the GM Telescope and Ion-Electron Detector (which had been removed for calibration by the experimenters). These units were later installed in place of the respective Flight Spare units for the purposes of shipment to Cape Canaveral. The Flight Spare Telescope had been removed for calibration while the Ion-Electron Detector was to undergo Acceptance Testing as well as calibration.

4. Solar Paddle shadow tests were conducted July 24 using the Flight Paddles in conjunction with the Prototype Unit. The unit was mounted on a variable-tilt spin table located out-of-doors, and spun at the orbital spin rate. The purpose of this test was to determine Paddle power output as a function of sun-angle. A discrepancy occurred when the turn-on plug was removed from the spacecraft thereby open circuiting all electronics from the power source, but the transmitter continued to operate for approximately one minute.

5. At the request of the State University of Iowa, the SUI CdS Sensors and Electron Spectrometer were switched from the Flight Spare to the Flight Spacecraft (and vice-versa). In addition, the readout and storage sequence of the SUI format was changed by a slight modification to the SUI Data Encoder.

6. Each of the experiments in the Flight Spacecraft and Flight Spare Spacecraft were given a final detailed calibration by the respective experimenters.

7. During a checkout (July 26) of the Flight Spacecraft's currents and voltages, the external power supply voltage was accidentally applied to the Transmitter thereby inducing a malfunction. The Transmitter was removed and replaced by the unit from the Flight Spare Spacecraft. In light of this critical situation, a second spare Transmitter was subjected to subassembly Acceptance Tests, sent to the Cape, and installed in the Flight Spare Spacecraft.

8. The thermal-coating was applied to the top and bottom covers (July 27) by the Mechanical Systems Branch.

9. It was determined on July 27, that the Optical Aspect sensors on all three spacecraft were somewhat defective. The Flight Spare and Prototype sensors were "non-flyable" because of defects on the silver-coated plates. The Flight Unit sensor, while acceptable for flight, was not completely free from defects. (It should be noted also, that this sensor was modified with respect to the sensitivity of one of the diodes after the completion of the test program but prior to delivery to the launch site.) It was surmised that at least some of the damage to the sensors had been the result of handling in transit.

10. A capacitor circuit was installed between the Magnetometer Electronics and the Telemetry Encoder. This installation was deemed necessary for the acquisition of data from the Magnetometer because of high background noise due to other spacecraft subsystems.

11. It was discovered that the black paint thermal-coating on the top cover would not adhere to the cover when the protective strip coating was removed.

12. Several Solar Paddle modules were replaced because of cracked cells or open circuits.

13. The Flight and Flight Spare Program Switch modules were removed from their respective spacecraft (July 30) in order to remove a capacitor from their circuitry. This capacitor was part of the original design and served to prevent spacecraft turn-on with no load on the system Regulator Converter. This condition could exist during the integration or test phase of the S-3 Program when experiments and electronics may not be installed or wired in the spacecraft. The Regulator Converter could be damaged if it was not loaded down properly and system turn on was effected. The capacitor in the Program Switch circuitry prevents turn-on under these conditions.

It was reasoned that since the S-3 program had progressed past the integration and test phases, the capacitor would serve no further useful purpose, and therefore should be removed. Furthermore, this single capacitor was an "in-line" item and there was no redundant circuit. The capacitor was removed and the cards reinstalled in the Flight Spacecraft.

During a system checkout (August 3), the Flight Spacecraft could not be turned off by command from the Blockhouse Control. The malfunction was traced to the Program Switch which was removed and replaced by the unit from the Flight Spare Spacecraft. However, the spare Program Switch malfunctioned in an identical manner. The Flight Program Switch was reinstalled in the Flight Spacecraft.

During checkout of the Flight Spacecraft (August 6) an additional problem with the Program Switch was encountered. Heretofor, the problems with the Program Switch were concerned with a marginal turn-off condition whereas the present difficulty concerned marginal <u>turn-on</u> capabilities. Investigation into the problem revealed that the marginal turn-on and turn-off condition resulted from the removal of the capacitor as described above.

Both the Flight and Flight Spare Program Switch units were removed from the spacecraft and the potting compound was removed so that a two-capacitor circuit could be installed. Two capacitors were installed in series, to increase reliability. No further problems were encountered with the Program Switch.

14. While preparing to calibrate the Ion-Electron Detector and Optical Aspect Sensor of the Flight Spacecraft (July 30) (in sunlight, solar paddles attached), the despin motors actuated. There was no apparent explanation of this occurrence. Efforts were made to actuate another set of motors under identical circumstances with hopes of establishing the cause of the detonations. These motors could not be made to actuate under the duplicate conditions.

On August 4, the Flight Spacecraft was delivered to the Douglas Aircraft Company balance facility. Prior to mating the Spacecraft to the X-248 motor, while the Solar Paddles were being electrically connected to the spacecraft, the despin motors again actuated. Once again, repeated efforts were undertaken to actuate the motors under identical circumstances. All conceivable causes were investigated. A complete review of the Spacecraft system wiring as well as the despin circuitry, was conducted. The exact cause of the detonations could not be pinpointed. However, several discrepancies in wiring design and procedures were located in the despin circuitry. The discrepancies, which included non-twisted leads and currents flowing in ground circuits, were corrected by modifying the despin circuitry.

15. The Flight Spacecraft was re-delivered to the balance facility (August 7). At this time, the Spacecraft weighed 84.08 pounds (including the protective strip coat,

sensor covers and other miscellaneous small items). The Spacecraft was mated to the X-248 Motor and the balance operations initiated on August 8 (F-5 day). The balancing was completed by August 10 and the composite Spacecraft/X-248 assembly was mated to the first and second stages of the Delta 6 launch vehicle.

16. The initial checkout of the spacecraft while "on the stand" was hampered by an unknown interference source. The interference apparently affected the Magnetometer in such an adverse manner that it in turn, disrupted all telemetry channels. The interference stopped after a few minutes when an overhead crane on the gantry was moved. The crane was then moved back to its original position but the interference did not reappear. No further interference problems were encountered.

17. Subsequent checkouts of the spacecraft on August 11 and 12 indicated that all experiments and instrumentation were functioning properly.

18. During the spacecraft checkout of August 13, it was ascertained that the telemetry signal exhibited an undesirable condition described as incidental frequency modulation.

This problem may have been present with this particular Transmitter (formerly the Flight Spare unit) for sometime - perhaps as long ago as Thermal-Vacuum Tests, and had been mistakenly attributed to a misadjustment of the Ground Station Tracking Filter. It was judged that this Transmitter was not acceptable for flight. Correspondingly, the Spacecraft/X-248 combination was separated from the launch vehicle, returned to the Douglas balance facility, and disassembled. The Transmitter was removed from the spacecraft and replaced by the Flight Spare #2 unit. (This FS#2 Transmitter had undergone subassembly Flight Acceptance Vibration and Thermal-Vacuum Tests.)

19. After the spacecraft had been reassembled, it was remated to the X-248 Motor, the composite assembly rebalanced and returned to the launch pad on August 14. Subsequent spacecraft checkouts were completed without difficulty.

20. Flight Day operations were commenced at 1310 EST on August 15, 1961. The countdown proceeded normally and the terminal count was begun at 2135 EST. No spacecraft problems were encountered throughout the countdown and the launch occurred at 2221 EST, August 15, 1961. The telemetry signal was monitored until T + 390 sec. at which time the vehicle passed over the radio horizon. At this time, all instrumentation and experiments were functioning properly.

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